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INTERIM FLIGHT REPORT ANCHORED INTERPLANETARY MONITORING PLATFORM AIMP I - EXPLORER XXXIII

BY

J. J. MADDEN

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ABSTRACT

The first Anchored Interplanetary Monitoring Platform was launched on July 1, 1966, into a highly eccentric earth orbit alternate mission instead of the proposed lunar orbit. The alternate mission was chosen because the over performance of the vehicle precluded a captured lunar orbit.

The orbital elements of the achieved orbit vary rapidly. In general, for the first six months the apogee will remain between 400,000 km and 530,000 km and the perigee between 30,000 km and 100,000 km. The closest approach to the moon (35,000 km) occurred on the initial orbit. Other close approaches (40,000 km to 60,000 km) occur in September, November and December of 1966.

The launch operations, orbit and spacecraft performance are discussed based on the first three months of data. Some predicted parameters are also included.

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INTERIM FLIGHT REPORT

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AIMP I - EXPLORER XXXIII

I. INTRODUCTION

The AIMP spacecraft is one of the IMP series of spacecraft designed to monitor interplanetary space. The initial intent was to place the AIMP spacecraft in an orbit about the moon. However, initial studies showed that even with all systems working within prescribed limits there existed at least a three out of ten chance that a lunar orbit might not be obtained. Alternate missions were studied that could best satisfy the scientific objectives. It was decided that a highly eccentric earth orbit having an initial apogee of 450,000 km and a perigee in excess of 30,000 km with a lifetime of at least 180 days would be acceptable for an alternate mission.

On July 1 the AIMP spacecraft was launched. Unfortunately, all the small errors associated with the launch vehicle, though within prescribed limits, were in the same direction resulting in a high-energy transfer trajectory. The fourth stage was fired six and one-half hours after launch to obtain the alternate mission orbit.

In this report, the launch phase and spacecraft performance will be discussed in detail. Other pertinent spacecraft data can be found by consulting the references and the appendices.

II. LAUNCH OPERATIONS

a. General

The AIMP spacecraft/third stage combination went on stand on June 22 in preparation for a June 30th launch date. All prelaunch tests planned for the spacecraft were performed satisfactorily. The final launch countdown commenced on F-2 June 28.

June 30th Launch Attempt—The spacecraft final prelaunch checks were made from 0259 hours to 0328 hours EST. The spacecraft was turned off at the end of these tests and the final arming of the fourth stage performed (0330 to 0340 EST). The weather conditions varied between heavy and medium rain. The

spacecraft was turned back on at approximately 0928 hours for a 1008.13 EST launch. The count proceeded with the weather conditions causing the situation to change from a go to a no go condition for vehicle guidance and tracking. The count went to T-3 at which time a hold for weather commenced. The window was extended by one minute (from 2 to 3 minutes). The count was reinitiated with the plan to go to T-0 and hold. The weather cleared sufficiently at the launch stand, however, an exceeding heavy downpour was experienced at the BTL Guidance radar locations. This plus the uncertainty of prevailing weather conditions along the vehicle trajectory coupled with insufficient test data of the effects on tracking of the additional antenuation of the radio waves due to heavy rain conditions caused BTL guidance personnel to request cancellation of the mission.

Sometime during the final count, the aircondition flexible duct extending from the service tower to the inlet hatch on the upper portion of the fairing broke loose at the service tower end (approximately 25 feet of duct work remained attached to the vehicle). The final filter which is within four feet of the fairing acted as an adequate block preventing excess moisture conditions in fairing. The duct was examined and found to be damp to a depth of about 5 feet from the break point. The duct was dried and secured. The hatches were removed and the spacecraft was found to be dry.

July 1 Launch—The F-0 day check was performed from 0353 to 0425 hours EST. The spacecraft had remained in the armed condition from the previous day since it was considered inadvisable to remove the plugs which were lock tighted into the spacecraft. The spacecraft was turned on at 1012 in preparation for an 1102:25 EST launch. The day was sunny with scattered clouds.

b. Launch Sequence

The sequence of events is given in Table I. The fourth stage events were initiated by ground command. The timers had to be started in order that a direct fire command could be used since all fourth stage events were interlocked with the command "timers on". The direct-fire command resulted in the firing of the fourth stage, reset of the two-hour timers, and switched the circuitry so that the count out of the timers resulted in separation of the fourth stage.

c. Summary Delta 39 Performance

Lift-off occurred at the opening of the window and the first stage started moving considerably left of nominal. This excursion was most likely due to misalignment of the solids with some reinforcement by the winds from the south

Table I

Event		Time (GMT)
Liftoff	7/1	1602:25.5
Solid Motor Separation		1603:34.9 (Visually observed)
Main Engine Cutoff		1604:56.9
Second Stage Ignition		1605:00.9
Fairing Jettison		1606:03.5
Second Stage Cutoff		1611:20.7
Spin-Up		1619:12.9
Stage 2/3 Separation		1619:24.7
Third Stage Ignition		1619:57.9
Third Stage Burnout		1620:26.9
Despin		1620:41.9
Paddles Deployed		1620:51.9
Separation		1621:21.9
Start Fourth Stage Timers		2200:00
Direct Fire Retro		2232:57
Retro Burnout		2233:13.2
Fourth Stage Separation	7/2	0033:47

after about 50 seconds the vehicle began to parallel the predicted trajectory. At 90 seconds vehicle ground guidance control brought the vehicle back to nominal and the remainder of first stage flight was uneventful. First stage performance was very close to predicted with thrust slightly low and propellent utilization slightly high. Early estimates indicate first stage velocity was within about 20 fps of nominal (14,500 fps) on the high side.

The second stage generally performed well; however, a combination of various guidance tracking errors, an error in shutdown impulse plus guidance logic limitations combined to effectively increase the energy at third stage burnout by about 0.2%.

The third stage imparted more energy to the spacecraft than nominally predicted.

The following Table II gives a comparison of the nominal values and those actually achieved.

Table II

Parameter	Units	D.T.O. Nominal	Launch Configuration Trajectory	Actual
Second Stage Cutoff				
Velocity (inertial)	ft/second	26,689.6	26,696.7	26,705.0
Inertial Flight Path Elevation Angle	Degree	0.33	0.27	0.29
Inertial Flight Path Azimuth Angle	Degree	95.94	95.97	95.94
Altitude	Nautical Miles	96.8	95.0	99.9
Energy/Mass	ft^2/sec^2	375,215,744	375,065,552	376,599,060
Radius at Apogee	Nautical Miles	4,220.9	4,218.60	4,254.4*
Radius at Perigee	Nautical Miles	3,537.4	3,535.86	3,540.3*
Inclination	Degrees	28.76	28.77	28.8*
Time	Seconds from Lift-off	540.0	540.0	546.625
Third Stage Cutoff				
Velocity (inertial)	ft/second	36,635.8	36,656.5	37,703.5
Inertial Flight Path Elevation Angle	Degree	2.07	1.85	2.08
Inertial Flight Path Azimuth Angle	Degree	111.20	111.30	112.30
Altitude	Nautical Miles	162.57	162.62	164.06
Energy/Mass	ft^2/sec^2	666,189,880	666,290,340	668,914,504
Apogee	Nautical Miles	299,836.3	350,935.8	468,207
Perigee	Nautical Miles	3,600.5	3,600.5	3,602.14
Inclination	Degrees	28.76	28.76	28.9
Time	Seconds	1,052.12	1,053.22	1,053.22

*Best Estimates

d. Retromotor Performance

The retromotor was a Thiokol TE-M-458 solid-fuel motor using an ammonium perchlorate polyurethane composite propellant. Physical characteristics of motor number 7 are as follows:

Weight of propellant	- 68.27 pounds
Weight of nozzle and case	- 9.95 pounds
Weight of two pyrogen igniters	- .68 pounds
Total weight	- 78.90 pounds

A performance analysis was made using the best estimates of the actual transfer trajectory and the best estimate of the final orbit based on the available tracking data. The results showed the maximum percentage of error for the thrust achieved to be approximately 0.6% (846.13 pounds nominal and 841.70 pounds actual). The temperature of the motor at the time of ignition was 28.5°C. Below is a tabulation of the nominal (based on tracking data up to ignition of fourth stage and nominal fourth stage performance) and actual parameters of the fourth stage:

	Actual Fourth Stage Burnout (1)	Actual Third Stage plus (1) Nominal Fourth Stage
Velocity (Ft/sec)	8915.59	8936.74
Flight Path Angle (Degrees)	50.86	50.77
Azimuth (Degrees)	82.24	82.43

(1) Measurements include errors due to inaccuracies in determination of orbital arcs.

The action time (includes most of motor tail-off) has a nominal value between 23.0 and 23.5 seconds (statistical sample available does not permit a more refined determination). This value was measured by an onboard "G" switch which gave a value of 23.2 seconds for the actual action time. Figure 1 gives the high temperature sensor plot for two hours following retromotor ignition. The starting value of 80°F agreed for both the high and low temperature sensors on the motor.

e. Spin Rate History

Table III below gives the nominal and measured values of the spin rate for the various launch events in RPM:

Table III

Event	Nominal	Measured by 3rd Stage Accelerometer	Measured from 136MC AGC Records	Measured by onboard OA System
Spin-up	149.2	141.0	140.8	
Separation 2/3 Stage		138.3		
Third Stage Ignition		135.6		
Third Stage Burnout		138.0	139.1	138.89
Despin	78.0	74.06	75.4	
Paddle Erection	41.3	39.59	41.1	
Boom Erection	27.5	26.10	26.9	26.76
Prior to 4th Stage burn				26.62
After 4th Stage burn				26.23
Prior to 4th stage separation				26.20
After 4th stage separation				26.25

In the spin-up six .6 KS40 and two .3 KS40 spin rockets were used. Had one .3 KS40 (.3 second burn time with a 40-pound thrust) failed the nominal spin rate would have been 139.4 rpm; therefore, from the measured results such a failure is a possible reason for the low initial spin rate. The spin down caused by third stage ignition is not at present explainable. Figure 2 gives the plot of the spin rate after despin to fourth stage separation.

f. Nutation

The third stage/spacecraft combination developed a 0.4° half-angle coning motion about the spin axis prior to ignition. The angle increased to a value of .76° during burning, having a rate of approximately 18 rpm. No cone angle could be measured with respect to the spacecraft after separation, which due to the limitation of the optical aspect system (cone angle measuring system) means any existing cone angle was less than .15 degrees.

g. Near-Real Time Control Center Operations

Transfer Trajectory Calculations—The tracking data taken at ETR during the launch phase was sent to GSFC in real time. The second stage burnout and third stage burn were not visible to ETR tracking systems. The early portion of the trajectory data indicated a near-nominal flight. In order to obtain spacecraft orientation, no ranging data was taken during the first twenty minutes after injection of the spacecraft into the transfer trajectory. The first available tracking data was from the Johannesburg Minitrack system. Interferometer system using 136.020 MC telemetry signal which results in measurement of the direction cosines to spacecraft from the station. This Minitrack data was used to compute a transfer trajectory and the resultant trajectory was a high-energy case indicating that an alternate mission would have to be chosen. This seemed in conflict with earlier results indicating a normal flight and it was considered inadequate data upon which to draw any conclusions.

The early range and range rate data from both the Tannarive and the Carnarvon tracking stations was erroneous due in one case to a false switch setting and in the other from tracking on a sidelobe. Immediate remedial steps were taken and the range and range rate stations began to operate properly. The inclusion of their first correct data showed that the spacecraft trajectory would not permit capture by the moon; therefore, at 4.5 hours after launch, it was determined that an alternate mission would be attempted.

The alternate mission fire times were run indicating that a fourth stage fire time of 2232 GMT would be required to meet the preset objectives of an apogee of 450,000 km and a perigee of 30,000 km. The detailed orbit study was run to determine lifetime, shadow conditions for first 180 days, and closest approach to the moon. The resultant orbit was found to be satisfactory and the fourth stage was fired at the prescribed time.

h. Near-Real Time Telemetry Reduction

Real time telemetry data was transmitted to GSFC from the KSC Satellite tracking station from T-35 minutes to loss of signal at approximately T+7 minutes. Ascension Island commenced sending real time data at approximately T+25 minutes. This was followed by data from the ships and the Kano, Nigeria station. All the telemetry data was processed satisfactorily. The orientation of the spacecraft was determined from the optical aspect system within 3 hours (telemetry data available within one hour but required orbit data was not available for approximately 3 hours) after launch. It was also determined that the spacecraft was not coning and that no serious coning or tipoff had occurred and the orientation was almost nominal (See table IV).

The spacecraft performance parameters were scanned and except for two minor anomalies were found to be nominal.

Table IV

Early Launch Spin Axis History

Item/Event	Nominal	After 3rd Stage Burn	After Boom Erection	After 4th Stage Burn	After 4th Stage Separation
Right Ascension	226.6°	227.6°	225.2°	225.2±1°	225.1±1°
Declination	-21.1°	-20.3°	-21.3°	-21.3±.8°	-21.3±8°
Spin Axis Sun Angle	130.5°	131.2°	129.2°	129.2°	129.2°
Coning Half Angle	0	<.15°	<.15°	≈.4°	≈.4°

i. Spacecraft Operation

Two anomalies were noted in the AIMP-D operation. The first was a failure of one telemetry binary performance parameter bit to indicate magnetometer booms locked in orbit configuration. This bit is controlled by two micro switches (one for each boom) connected in series. Thus, failure of either switch to close would cause the bit to remain a one indicating booms open. The initial design of the mechanical system (a long plunger extended through spacecraft bottom platform which caused a micro switch attached to an internal support strut to be forced into the closed position when the booms locked in against the bottom platform) had the tolerances set extremely close so that the micro switch did not show booms locked until the booms were essentially flush with the spacecraft platform. Any slight change of the spacecraft platform or micro switch bracket and plunger assembly due to paddle and boom erection could have caused a failure in the operation of one of the switches. At fourth stage firing the bit changed from a one to a zero (booms locked) and remained in this condition until fourth stage separation when it returned to zero. This indicates a marginal micro switch closure condition. The spin rate showed that the booms had been properly extended during the initial deployment. The retro fire would have assured locking them in position had there been a marginal case with respect to the booms. On AIMP-E each boom will be monitored separately and the micro switch system mechanical tolerances will be improved.

The second anomaly was with respect to the binary performance parameter bits used to monitor whether the fourth stage timers are running. Four bits are used — two for each of the redundant timers. The condition of these bits is

controlled by flip-flops attached to an output from the first and second decades (one bit each). The bits come up in an arbitrary state when the power is applied to the fourth stage timer system. Once the system is on, the bits should remain stable unless the timers are started and then they will change state each time a pulse is generated by the decade being monitored. Power is applied to this system prior to launch. During the third stage separation sequence (paddles, booms up) the bit configuration changed state. At this time both Iowa and MIT experiments are turned on. The bit configuration remained stable until fourth stage timers were started approximately five hours later. It is presently thought that the surge reflected back to the primary spacecraft power accompanying the Iowa/MIT turn on caused the bits to change state. On AIMP-E, additional filtering will be added to prevent the reoccurrence of this phenomenon.

No other discrepancies were noted in the AIMP-D operation during launch and early trajectory phase.

III. SPACECRAFT PERFORMANCE IN ORBIT

a. Spacecraft Operation

The spacecraft operation, except for an anomaly in the optical aspect system, minor excersion in temperature above predicted values, and a partial failure of California experiment; has been nominal. There has not been any significant degradation of the instruments noted in this three month period.

Optical Aspect System Anomaly—The spin period as telemetered is the number of eight hundred cycle counts occurring between two successive sun pulses. Occasionally in the telemetry space designated for the spin period a number equivalent to approximately 12 milliseconds appears. The 12 milliseconds corresponds to the length of a normal sun pulse as determined in spacecraft testing. It is at present thought that a noise pulse on the trailing edge of the sun pulse occasionally triggers the circuit (appears to be the next sun pulse leading edge) thus giving a measurement of sun pulse width. The triggering circuit is sensitive to noise during the time of the trailing edge of the sun pulse. The phenomena will be examined over the life time of the spacecraft to determine if this deduction is a correct one. The sun pulse width varies with temperature, however, if a long shadow is not encountered, the temperature variation may not be sufficient to cause a change in sun pulse width. The phenomena has existed since launch, however, it was not noticed until the spacecraft was in the orbit mode. The time interval between the appearances of this abnormal reading is normally several hours. It is felt that this abnormal condition has always existed. Due to the limited testing condition, the abnormality was not evidenced during prelaunch ambient and environmental examinations of the system.

On 23 August, the California experiment began to encounter periods of abnormal behavior. A few days later the two Geiger tubes showed all zeros on each readout. It is at present thought that one of the Geiger tubes has gone into continuous discharge. This would cause the voltage to drop below the starting value required for the other Geiger tube but would still be high enough for correct operation of the ion chamber. The continuous discharge could have been imposed in some way by exposure of the thin window G-M tube to the sun. Further investigation is necessary. It is doubtful that additional data will be available unless the spacecraft is commanded or goes into undervoltage condition. The ion chamber continues to operate satisfactorily.

b. Spin Axis Sun Angle and Spin Rate

The spin axis sun angle was approximately nominal at the insertion into the final orbit and the value of the angle has followed predicted values (See Figure 3). Table V contains a list of right ascention and declination of the spin axis as computed from the optical aspect data.

Table V

AIMP-D Spin Axis Position
(Referenced to Mean Equator and Equinox 1950.0)

Day of Year		Right Ascension	Declination
1 July	181.6	225.0	-21.3
	185.0	224.7	-21.5
	190.0	224.4	-21.8
	195.0	224.2	-22.0
	200.0	223.9	-21.9
	205.0	223.6	-21.8
	210.0	223.2	-21.8
	215.0	223.0	-21.7
	220.0	222.6	-21.6
	225.0	222.4	-21.5
	230.0		
	235.0		

Computed October 20, 1966.

The initial slow down in the spin rate is inconsistent with prediction since the sun is shining on the bottom of the spacecraft and should cause a spin-up. The rate of increase in spin rate should equal the rate of spin down, i.e., the curve should be symmetrical about the spin axis sun angle 90° point. The difference in slope of the spin up portion of the curve and the initial spin down are thought to be caused by outgassing of the spacecraft, i.e., loss of mass from the center of spacecraft.

c. Performance Parameters

There are twenty-six analog performance parameters.

- PP-1 12 Volt Buss. The value of the voltage monitored has remained between 11.9 and 12.0 volts since launch. This is within the acceptable 1% limit.

- PP-2 Battery Voltage. The battery voltage from lift off till July 2 at 0528 hours except for the short shadow period when the spacecraft went on battery power remained at 19.6 volts indicating the battery on the high charge rate. On July 2 the battery voltage readings changed to 18.3 where it has remained. Battery voltage readings during the shadow period are scarce due to the high amount of range and range rate data being accumulated at this time, however, it appears that the voltage drop to a low reading of about 14 volts during this period.

- PP-3 Battery Current. The battery current sensor that measures battery input or output current from 100 to 200 milliamps. Its prime purpose is to monitor the battery final charging rate prior to the switch from the 19.6 volts to the 18.3 volts. The switch point occurs when the battery is approximately 90% charged. The sensor will be saturated and read 200 milliamps if the battery is either in a high state of discharge thus accepting all excess current from the solar array or when the battery is supplying the spacecraft power. The plot of the battery charge current is given in Figure 4 for both charge cycles so far experienced in AIMP-D. Due to noisy data, the exact length of the battery operation during shadow cannot be determined from spacecraft data.

- PP-4 Solar Array Current. The solar array current varies with sun angle and orientation of the spacecraft to the sun. The current reading has varied between 3.0 and 5.0 amps during this time period. The 3.0 amps is the lowest reading experienced and is 0.9 amps above normal spacecraft

loads. It is estimated that the solar paddle output has degraded less than 5% during this time period.

- PP-5 Spacecraft Current. Normal spacecraft current readings have been 1.9 amps at 19.6 volts and 2.0 amps at 18.3 volts for average loads and 2.0 at 19.6 volts and 2.1 amps at 18.3 volts for normal peak loads. The current is 2.3 amps when Ames flipper is energized and 2.4 amps when the GSFC flipper is energized. The Ames flipper power is on for ten minutes (timed by an encoder pulse) and the GSFC flipper power for approximately five minutes (Controlled by a micro switch cut off at end of flip cycle but limited to maximum of the ten minute encoder pulse) as measured by telemetry.

- PP-6 28 Volt Buss. The value has remained approximately 28.3 volts from liftoff.

- PP-7 7 Volt. This voltage use for the thermistors has remained within limits since launch, i.e., either 7.0 or 7.1 volts.

- PP-8, 9, 10, 11, 12 Iowa Voltages and Solar Cell Damage Experiment. This data is not reported on in this document.

- PP-13, 14, 17, 18, 19 and 21 through 26 Standard Temperature Measurements. In general, the AIMP spacecraft thermistors by use of compensating networks were made to fit the same calibration curve thus simplifying data processing. The following is a list of the temperatures monitored along with the corresponding figure number on which the data is plotted.

Temperature Monitored	Figure Number
PP-13 Solar Cell Damage Experiment	6, 13
PP-14 Fourth Stage Low Temperature/Thermal Ion Temperature	6, 8
PP-17 Transmitter Temperature	5, 10
PP-18 Battery Temperature	5, 11
PP-19 Prime Converter Temperature	5, 12
PP-21 Ames Boom Temperature	13
PP-22 Ames Electronics Temperature	6, 14
PP-23 GSFC Boom Temperature	15

Temperature Monitored	Figure Number
PP-24 University of California Temperature	6, 16
PP-25 MIT Temperature	6, 17

The temperatures in general are satisfactory for the operation of all instruments and experiments, however; the battery temperature is excessive to that planned and will shorten its useful lifetime. Fortunately there are few shadow conditions predicted and these conditions are of short duration (See Table VI) and occur in the early life of the spacecraft.

Table VI

Shadow Predictions for AIMP-D
July 1966 to August 1968

Shadow	Entrance	Duration Hours	True Anomaly
12/22/66	12:13:20	.882	316°
1/6/67	00:42:10	.693	330°
1/18/67	12:17:10	.378	350°
2/1/67	22:59:22	.544	8°
2/14/67	09:46:18	.883	31°

The reason most of the temperatures exceed or approach the upper limits predicted when the sun is shinning on the top cover is not explainable at the present time. The temperature indicates that the top surface of the spacecraft has been contaminated. The source of this contamination is not identifiable, however, there seems to be four potential sources which are: (1) the vehicle fairing (predicted heating from the actual trajectory flown seems insufficient to produce outgassing), (2) blow back from the fourth stage, (3) outgassing of the spacecraft, and (4) micrometer damage of the buffed aluminum surfaces. It is doubtful that the exact source of the decontamination can be determined, however; if the bottom surfaces (particularly those of the booms) indicate the same type of degradation when the sun shines on them the second time, then it is doubtful that (1), (2) and (3) of the above are the major sources of this contamination. The original measurements of temperatures on the bottom are also slightly high in most instances.

- PP-15 Fourth Stage High Temperature/Fourth Stage Firing Duration. The quantity measured by this performance parameter is switched from the fourth stage temperature to the duration of burn upon the separation

of the fourth stage. A plot of the temperature during the fourth stage burn and cool down is given in Figure 1. A reading from the low temperature sensor is given for reference. The low temperature sensor was destroyed shortly after ignition. The temperature profile is within the range of expected values; however, due to the number of variables involved no exact predicted curve is given. The temperature was predicted to peak between 600 and 750 degrees F and to taper off slowly because the thermal blanket retains the heat. The measurement of the duration of thrust was 23.2 seconds.

- PP-16 Encoder Temperature and Calibration. PP-16 is subcommutated within the encoder to obtain the following: two temperature readings, two readings of ground, a 4 volt standard reading, a 4 volt divider reading, a 2.5 volt reading and 5 volt reading. Both temperature readings have remained identical and a plot of one of them is given in Figure 9. Both ground readings had a decimal value of 220 on launch day and remain at this value till July 26 when at an encoder temperature of 17°C the reading changed to 219. It remained a 219 until August 29 when at an encoder temperature of 19°C it returned to 220 where it remained for the period (July 1 to September 30) covered in this report. The 4 volt standard and 4 volt divider readings have varied between 60 and 61 from launch to July 27 when at an encoder temperature 16°C they both tended to stop varying taking on the steady value 60. They remained in this status till September 1 when at an encoder temperature of 21°C the values started to vary between 60 and 61 with a tendency as the temperature increased to stay at a value of 61. The 2.5 volt and 5 volt readings have not varied since launch having the values 121 and 19 respectively.

- PP-20 Solar Array Temperature. Due to the extreme temperatures this thermistor sees, it is not standardized; thus, requires its own calibration curve. Two plots of this temperature are given. One covers the first (Figure 19) shadow period experienced by the spacecraft, and the other the temperature covers the period (July 1 to September 30) Figure 13. In Figure 19 the temperature of the solar cell experiment is plotted for reference since it also reacts fairly rapidly to a shadow condition.

The temperature characteristics of the solar array are extremely hard to predict because of the variation in individual solar cells.

IV. AIMP-D ORBIT

Orbital Characteristics

The AIMP-D orbit is highly perturbed by the moon. This effect of the moon is accentuated during close approaches. Examination of Figures 20, 22 to 25, will show the effect on the orbit of the first lunar close approach. Further examination of Table VII will also indicate similar, though not as drastic, changes since the first orbit had the closest approach to the moon (36,000 KM). Table VII gives the orbital elements from July 1966 to August 1968. Initial checks indicate that the prediction program is accurate for a period of at least six months.

It should be noted that the moon perturbation adds or subtracts energy to the orbit. This is best indicated by the variation in the period of the orbit. The initial period was approximately 15 days and varies up and down reaching a low of approximately 12 days in the latter part of January 1967 from which, by staggered steps, it reaches a value of approximately 30 days by August of 1968. It is also evident from the table that the orbit tends to become circular.

The spacecraft line of apsides-sun angle has gone from 116.6 to 151.1 during the time period 1 July to 30 September (see Figure 26).

Table VII

Orbital Elements for AIMP-D
July 1966 to August 1968

Date	Perigee (KM) Radius	Apogee (KM) Radius	Inclination	Closest Approach to the Moon (KM)
7/8		440,000*	7.0	
7/13	50,000		7.2	
7/21		482,000*	7.4	
7/29	47,000		7.5	91,500*
8/2		497,000	12.0	
8/15	66,000		14.4	
8/24		514,000*	14.0	
9/2	65,600		13.8	
9/11		514,000	14.2	

Table VII (Continued)

Date	Perigee (KM) Radius	Apogee (KM) Radius	Inclination	Closest Approach to the Moon (KM)
9/20	62,800		14.1	
9/26			25.3	60,000
9/30		474,000	22.2	
10/6	83,000*		21.8	
10/14		484,000	21.2	
10/23	96,500		20.5	
11/1		477,000	20.7	
11/10	94,200		20.7	
11/18		456,000	22.2	
11/20			23.2	59,700
11/25	49,000		21.7	
12/2		458,000	22.0	
12/10	45,400		22.1	
12/16			24.0	51,500
12/16		443,000	24.0	
12/22	36,900		27.7	
12/29		464,000	27.7	
<u>1967</u>				
1/6	32,200		28.0	
1/13		449,000	34.7	58,000
1/18	32,400		34.8	
1/25		464,000	35.1	
2/1	32,400		35.4	
2/7		445,000	41.9	55,700
2/14	44,200		43.2	
2/21		452,000	43.3	
2/28	48,000		42.8	
3/8		432,000	51.2	55,000
3/14	97,000		48.6	
3/22		438,000	48.5	
3/30	104,000		47.5	
4/7		434,000	47.6	
4/15	103,000		47.7	
4/23		441,000	47.6	
4/29			47.6	286,000
5/1	103,000		47.6	

Table VII (Continued)

Date	Perigee (KM) Radius	Apogee (KM) Radius	Inclination	Closest Approach to the Moon (KM)
5/10		448,000	48.2	
5/18	92,000		48.6	
5/25		436,000	50.4	
5/27			49.4	46,000
6/1	105,000		41.8	
6/10		494,000	41.8	
6/20	113,000		41.7	
6/29		487,000	41.7	
7/9	109,000		42.2	
7/17		460,000	42.4	
7/20			43.8	72,000
7/26	144,000		38.7	
8/6		505,000	38.4	
8/17	168,000		37.8	
8/29		521,000	37.3	
9/9	172,000		36.6	
9/21		524,000	36.7	
10/2	149,000		36.0	
10/5			35.9	281,000
10/12		513,000	35.6	
10/23	131,000		36.3	
10/24			36.3	250,000
11/3		501,000	36.3	
11/6			36.4	144,000
11/12	143,000		35.6	
11/24		537,000	35.4	
12/3			35.6	175,000
12/6	148,000		35.4	
12/18		532,000	35.1	
12/30	175,000		34.4	
1968				
1/12		534,000	34.1	
1/24	198,000		32.9	
2/5		516,000	32.5	
2/17	197,000		31.6	
3/1		490,000	31.4	

Table VII (Continued)

Date	Perigee (KM) Radius	Apogee (KM) Radius	Inclination	Closest Approach to the Moon (KM)
3/11	173,000		31.3	
3/16			29.8	175,000
3/24		489,000	29.7	
4/2	182,000		30.0	
4/12		494,000	29.9	
4/16			30.3	108,000
4/23	168,000		26.4	
5/6		553,000	26.2	
5/14			26.0	161,000
5/19	165,000		24.5	
6/3		565,000	24.0	
6/11			23.5	290,000
6/15	188,000		23.4	
7/1		580,000	22.6	
7/11			21.4	272,000
7/14	225,000		21.3	
8/1		591,000	20.5	
8/11			16.1	129,000
8/13	262,000		16.0	

ACKNOWLEDGMENT

The following personnel contributed data or information contained in this report. Messrs. D. McCarthy, S. Ollendorf and D. Miller (Westinghouse employee). Messrs. T. Flatley, J. Albus, W. Mish, and T. Hollis contributed spacecraft data. Messrs. W. Schindler and R. Goss the DAC vehicle Data. Messrs. W. Bryant and J. Barsky orbital data.

The report was compiled by the AIMP project office with the aid of Mr. W. Lumadue and Mrs. D. Matters.

REFERENCES

(1) AIMP Summary Description, NASA X-672-65-313, 1965.

(2) E. W. Travis and D. K. McCarthy, AIMP-D Mechanical Design, Development and Testing, NASA X-723-66-419, 1966.

(3) D. L. Miller and E. W. Travis. The AIMP-D Launch Operations Log, NASA X-723-66-472, 1966.

(4) Flight Report for Delta Vehicle S/N 467/20207/20205 Delta Program - Mission No. 39 spacecraft, AIMP-D. Douglas Report SM-52424, 1966.

APPENDIX A

SPACECRAFT DATA

The following tables and diagrams list or illustrate the final measurements made on the AIMP-D prior to launch

Voltage and Current Table

Unit	Supply	Current
Transmitter	+28V	610 MA
Total	+28V	610 MA
MIT	+20V	85-180 MA
OA	+20V	60-106 MA
Total	+20V	145-240 MA
Ames	+12V	57 MA
GSFC	+12V	92 MA
Iowa	+12V	58 MA
U/Cal	+12V	11 MA
I&E	+12V	127 MA
Telemetry	+12V	95 MA
Encoder	+12V	90 MA
Performance Parameter	+12V	8 MA
Fourth Stage Electronics	+12V	10 MA
Fourth Stage Parameter	+12V	7 MA
TOTAL	+12V	555 MA
Spacecraft	+18.24V	2,000 MA

AIMP-D Serial Number Designations and Final Weight

Experiment or Instrument	Flight Spacecraft Serial Number	Flight Spacecraft Weight (Lbs.)	
Ames Sensor EA1	02	1.04	Total
Ames Data Handling EA2	02	1.20	Ames
Ames Signal Processor EA3	02	1.41	
Ames Sensor Electronics EA4	02	1.54	5.19
U/Cal EC1	01	1.53	Total
GSFC Sensor EG1 (04) & Flipper (01)	04	1.17	GSFC
GSFC A/D Electronics EG2	02	1.96	Mag.
GSFC F/G Electronics EG3	04	1.68	4.81
U/Iowa EI	02	2.12	
MIT Sensor EM1	03	3.92	Total
MIT Logic #2 EM2	03	1.43	MIT
MIT Logic #3 EM3	03	1.52	6.87
GSFC Thermal Ion Experiment ES1	02	4.08	
Encoder ID1	02	4.44	
Performance Parameter ID2	02	0.89	
Transmitter IT1	02	1.51	
R&RR #1 IT2	02	1.35	
R&RR #2 IT3	02	1.30	
R&RR #3 IT4	02	1.28	
Decoder #2 IT5	02	1.31	
Command Receiver #2 IT6	02	1.18	
Antenna Cups IT7	03 #'s 5, 6, 7, 8	.49	
Antennae		.2	
Hybrid Card IT8	02	.58	
Solar Array Regulator IP2	02	.43	
Battery IP3	05	10.00	
Prime Converter IP4	02	3.73	
Optical Aspect Converter IP5	02	.56	
Encoder Converter IP6	02	.83	
Undervoltage IG1	03	1.04	
Fourth Stage Electronics IG2	02	1.23	
Flipper Control IG3	03	.46	
Optical Aspect Sensor IA1	02	.89	
Optical Aspect Amplifier IA2	03	1.38	
Optical Aspect Computer IA3	04	1.19	
Solar Cell Experiment IH4	02	.22	

AIMP-D Serial Number Designations and Final Weight (Continued)

Experiment or Instrument	Flight Spacecraft Serial Number	Flight Spacecraft Weight (Lbs.)
Solar Paddles	06, 07, 08, 09	21.7
Electrical Harness		9.00
Structure		33.18
Fourth Stage Motor	07	78.90
Fourth Stage Attach Hardware		2.56
Nutation Dampers		.69
Despin Weights		.30
		207.42

Moments of Inertia

Configuration	Weight (Pounds)	C.G. Along Z Axis (Inches)	Ixx Slug-Ft^2	Iyy Slug-Ft^2	Izz Slug-Ft^2
Launch (All appendages folded)	207.42	8.21	10.43		3.65
Yo-Yo Deployed	207.42				
Paddles Erected	207.42	10.42	8.88		9.18
Booms Erected	207.42	11.17	6.99	13.36	15.06
Post Retro-Fire	137.85				
Post Retro-Separation	125.75	5.42	4.35	10.74	14.72

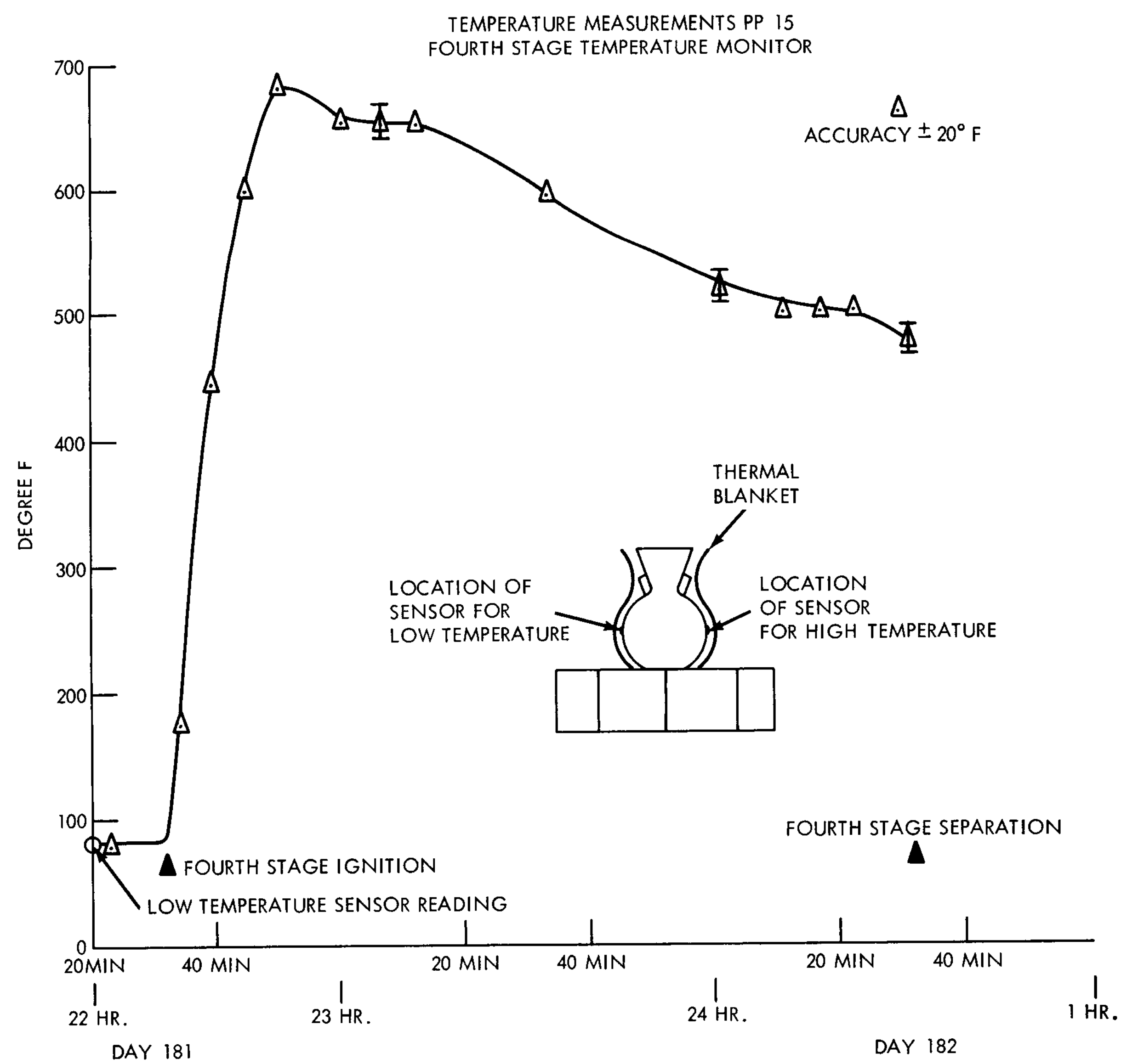


Figure 1—Temperature Measurements PP 15, Fourth Stage Temperature Monitor

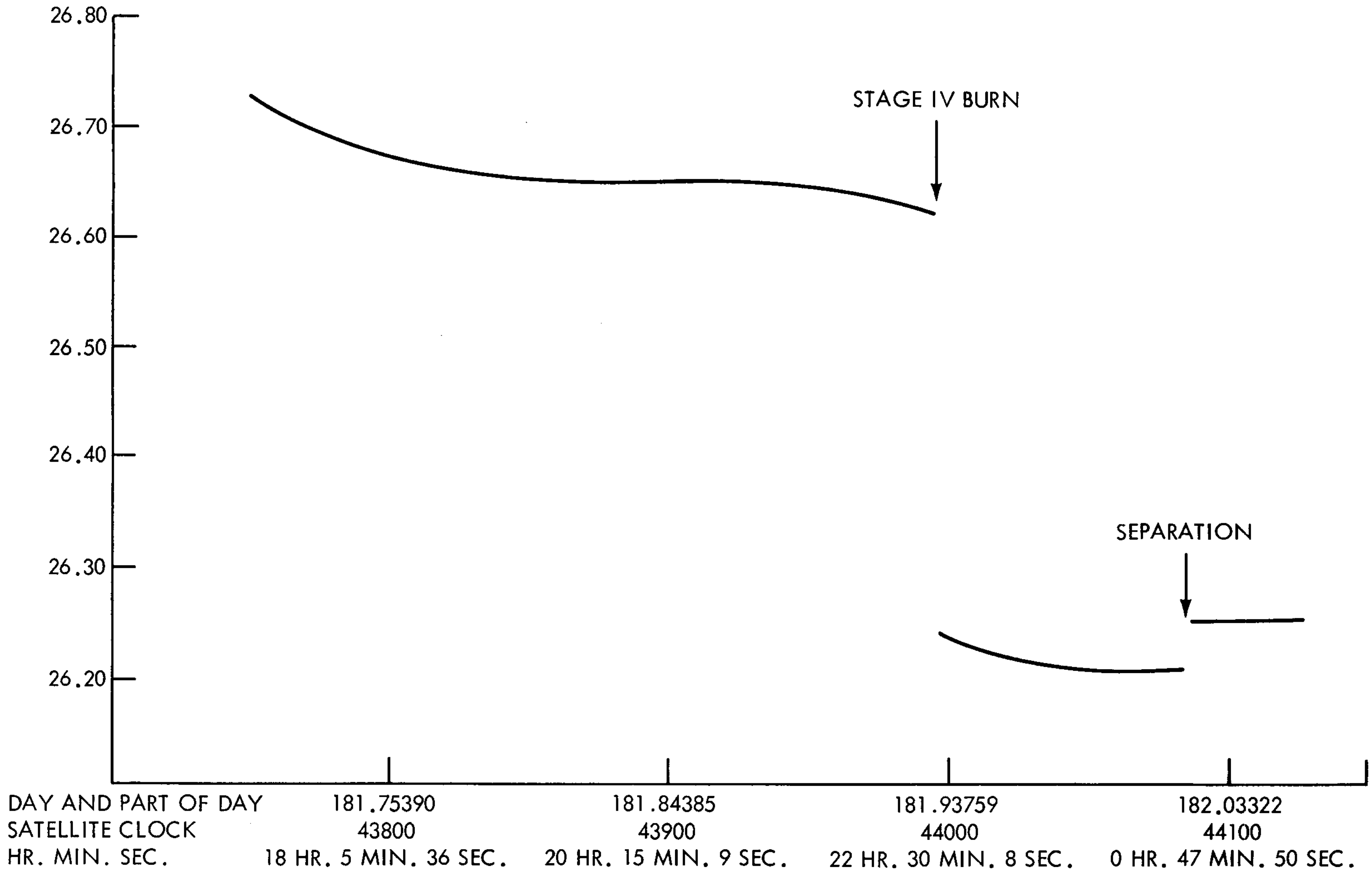

26.80
26.70
26.60
26.50
26.40
26.30
26.20
STAGE IV BURN
SEPARATION
DAY AND PART OF DAY
SATELLITE CLOCK
HR. MIN. SEC.
181.75390
43800
18 HR. 5 MIN. 36 SEC.
181.84385
43900
20 HR. 15 MIN. 9 SEC.
181.93759
44000
22 HR. 30 MIN. 8 SEC.
182.03322
44100
0 HR. 47 MIN. 50 SEC.

Figure 2

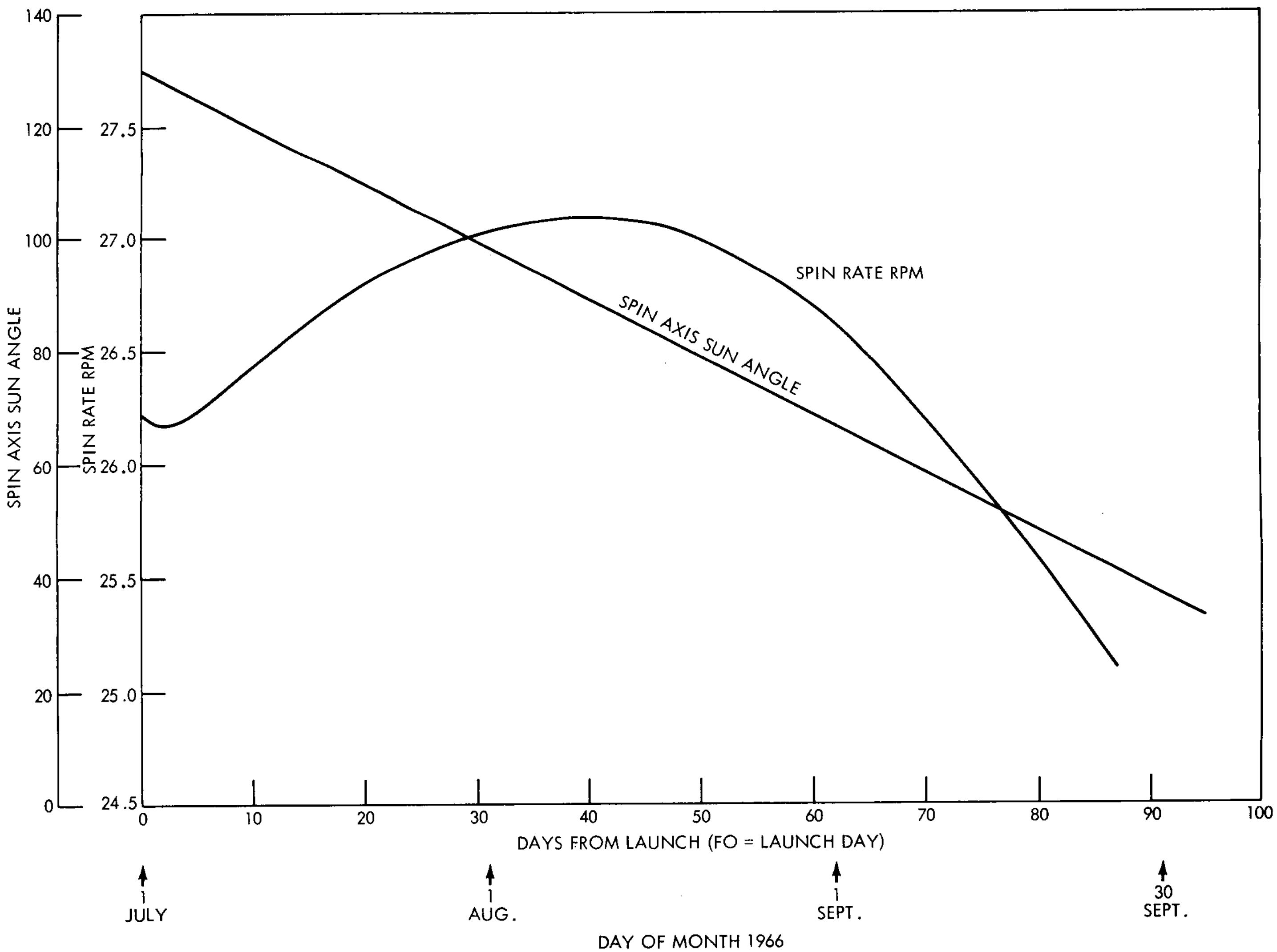

SPIN AXIS SUN ANGLE
SPIN RATE RPM
140
120
100
80
60
40
20
0
27.5
27.0
26.5
26.0
25.5
25.0
24.5
SPIN RATE RPM
SPIN AXIS SUN ANGLE
0
10
20
30
40
50
60
70
80
90
100
DAYS FROM LAUNCH (FO = LAUNCH DAY)
1
JULY
1
AUG.
1
SEPT.
30
SEPT.
DAY OF MONTH 1966

Figure 3

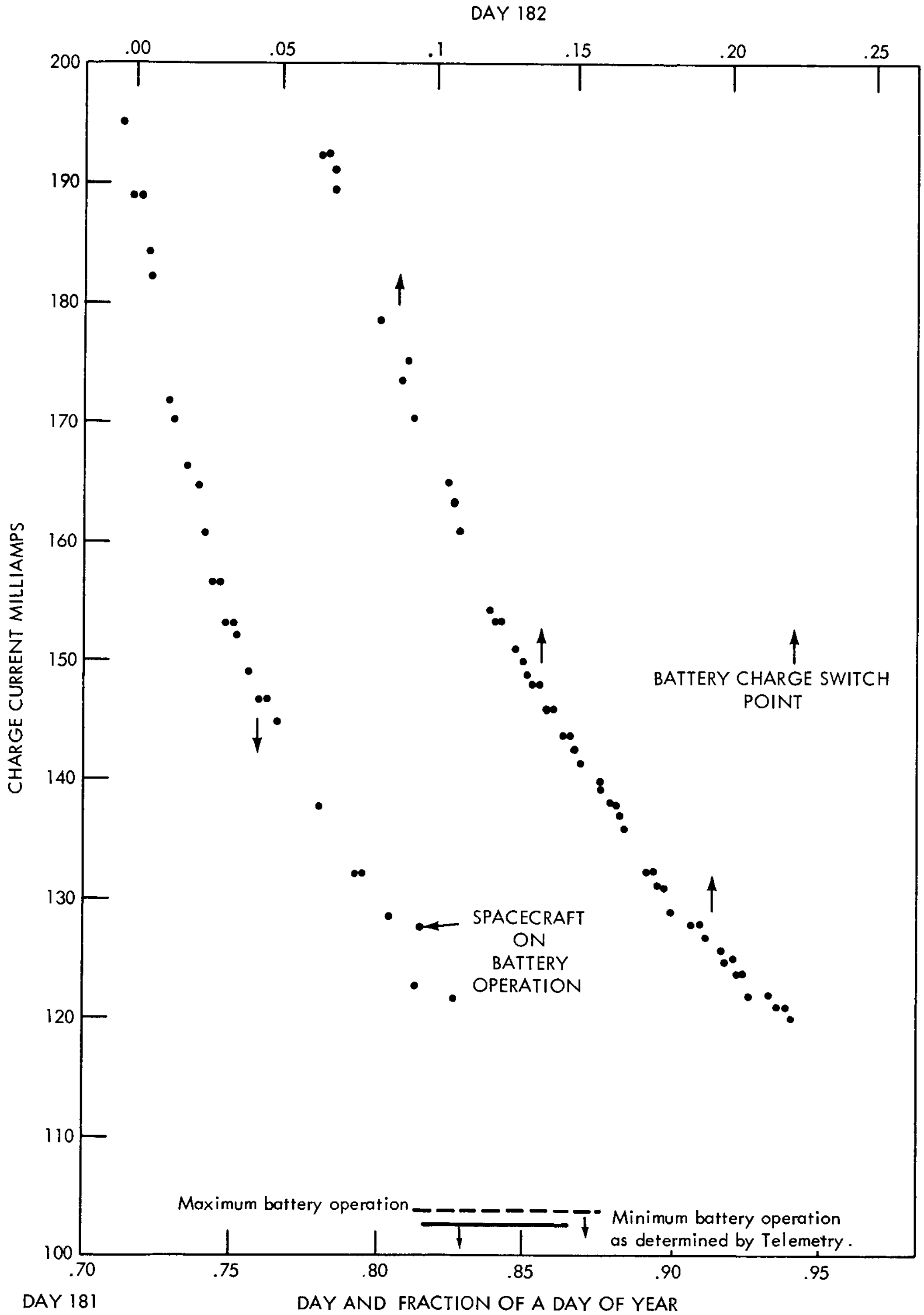

DAY 182
.00
.05
.1
.15
.20
.25
200
190
180
170
160
150
140
130
120
110
100
CHARGE CURRENT MILLIAMPS
BATTERY CHARGE SWITCH POINT
SPACECRAFT ON BATTERY OPERATION
Maximum battery operation
Minimum battery operation as determined by Telemetry.
.70
.75
.80
.85
.90
.95
DAY 181
DAY AND FRACTION OF A DAY OF YEAR

Figure 4

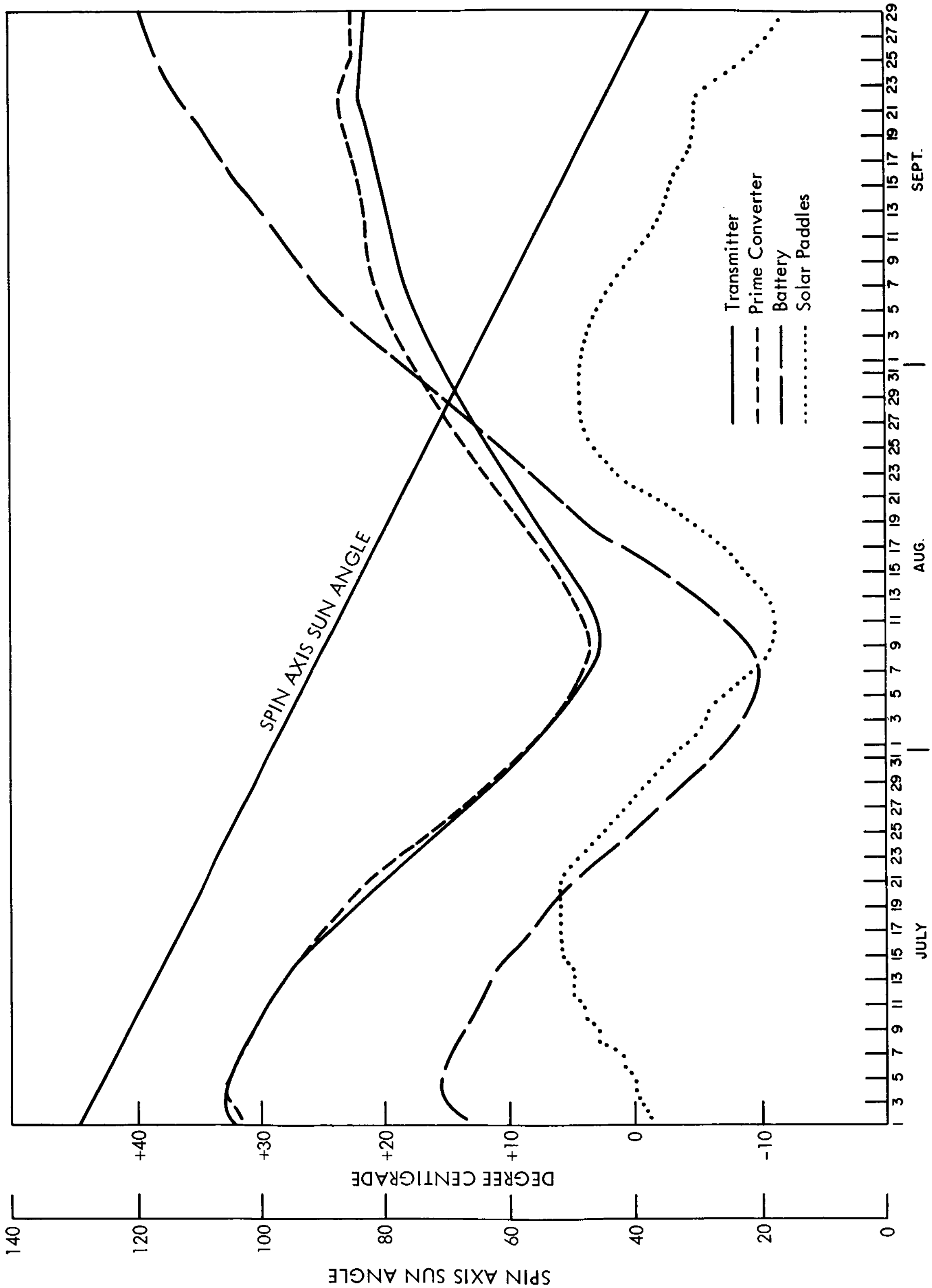

SPIN AXIS SUN ANGLE
Transmitter
Prime Converter
Battery
Solar Paddles
DEGREE CENTIGRADE
+40
+30
+20
+10
0
-10
SPIN AXIS SUN ANGLE
140
120
100
80
60
40
20
0
JULY
AUG.
SEPT.

Figure 5

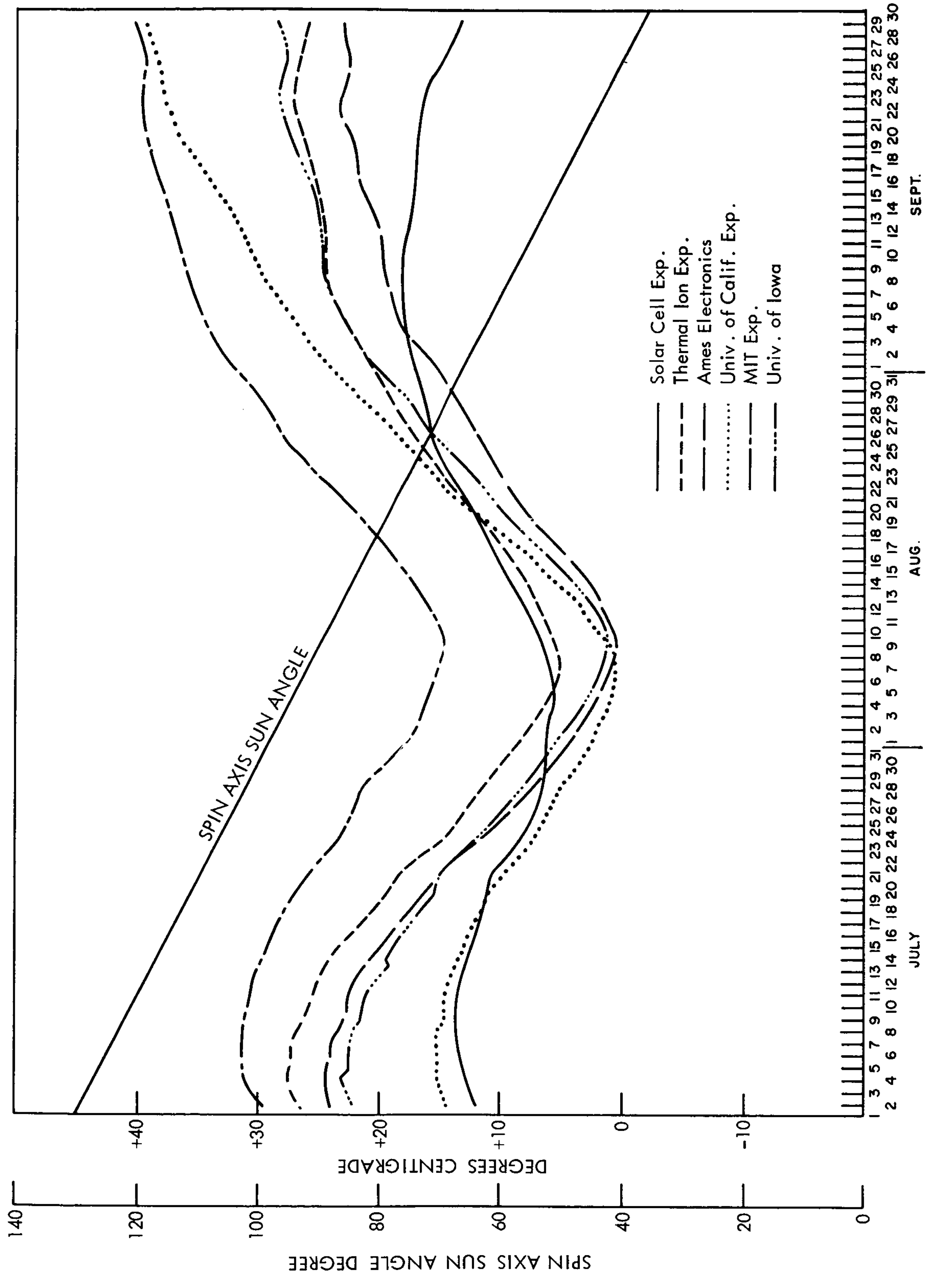

SPIN AXIS SUN ANGLE
Solar Cell Exp.
Thermal Ion Exp.
Ames Electronics
Univ. of Calif. Exp.
MIT Exp.
Univ. of Iowa
DEGREES CENTIGRADE
+40
+30
+20
+10
0
-10
SPIN AXIS SUN ANGLE DEGREE
140
120
100
80
60
40
20
0
JULY
AUG.
SEPT.

Figure 6

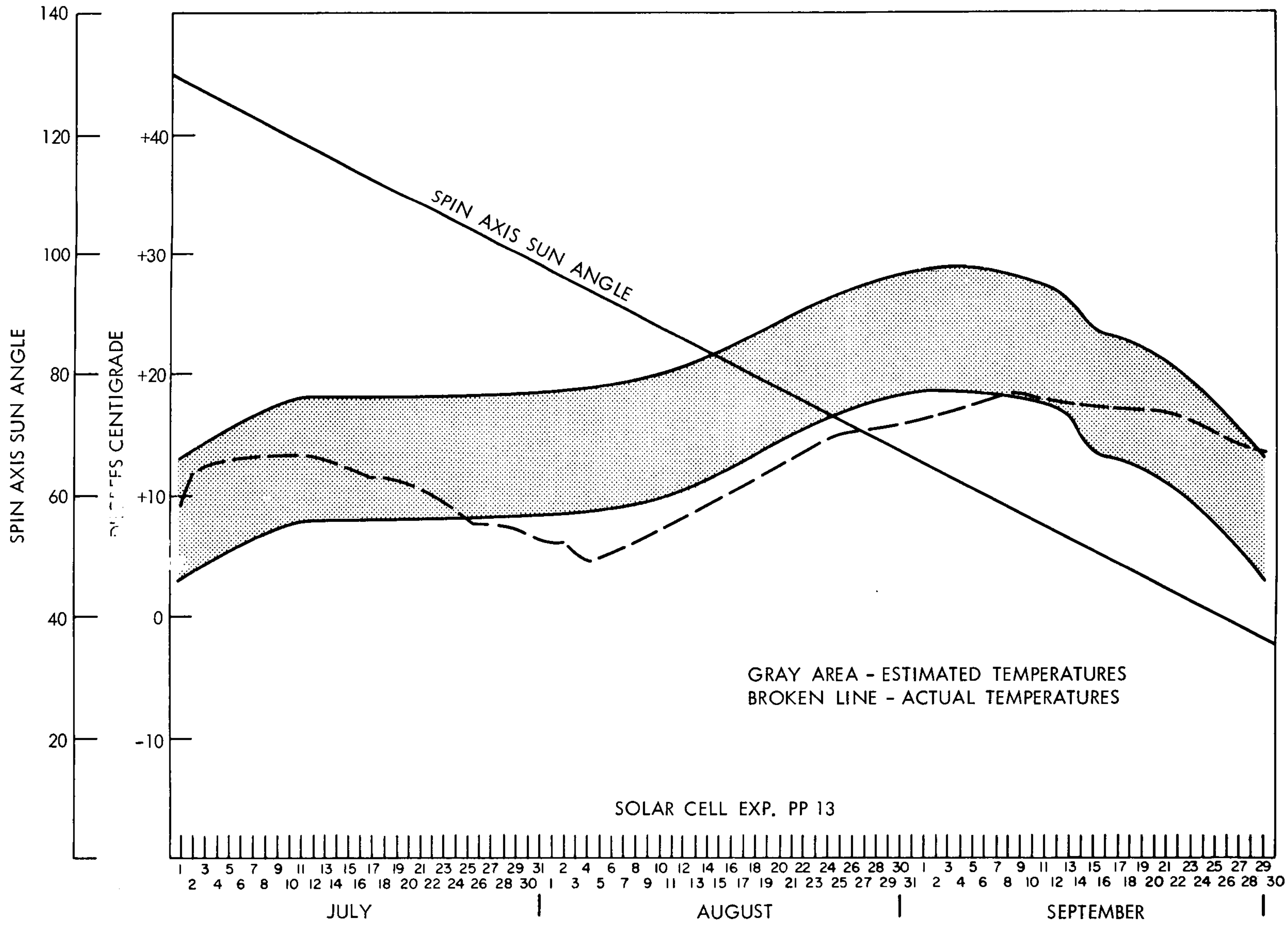


Figure 7– Solar Cell Experiment PP 13

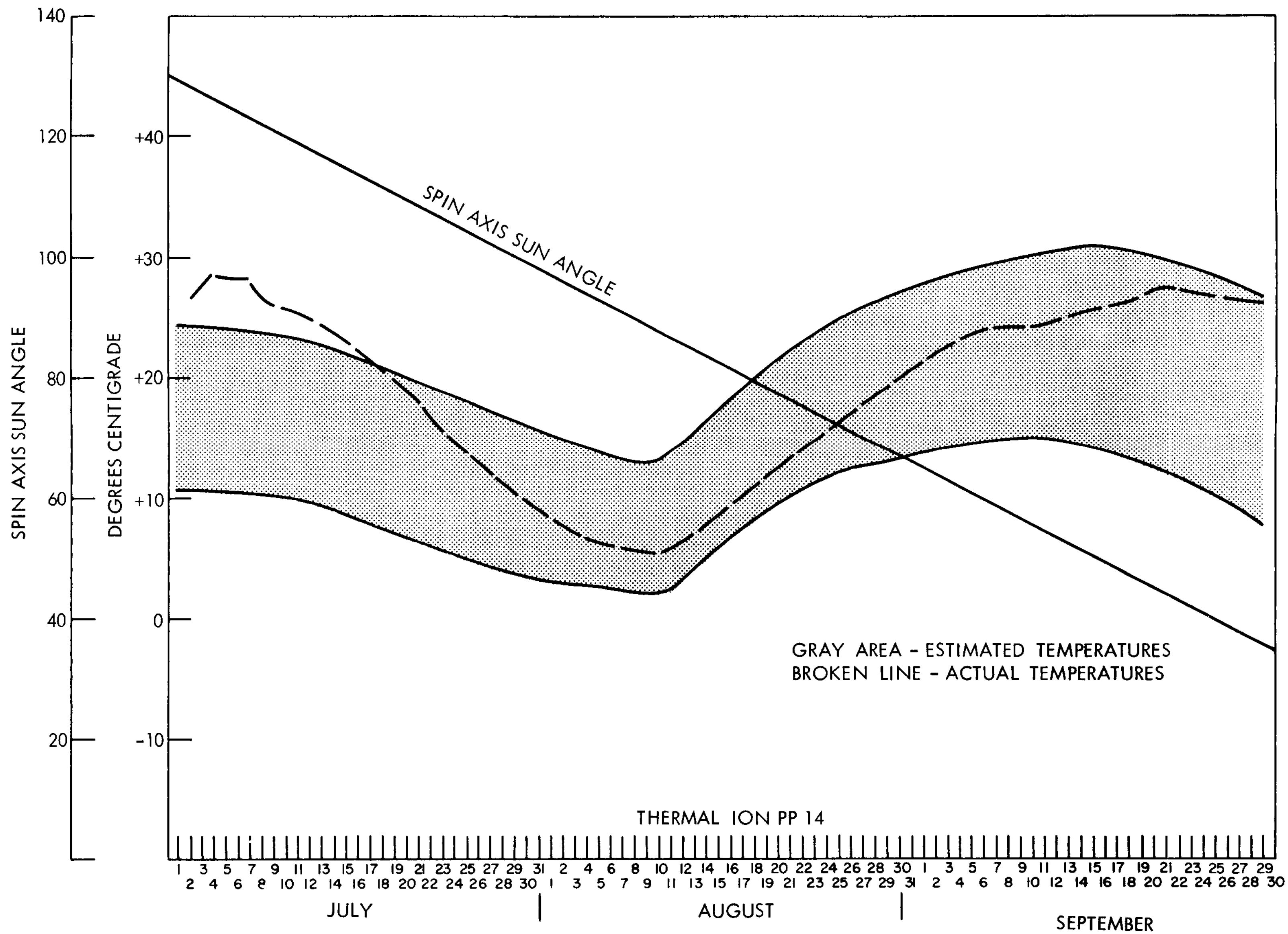


Figure 8–Thermal Ion Experiment PP 14

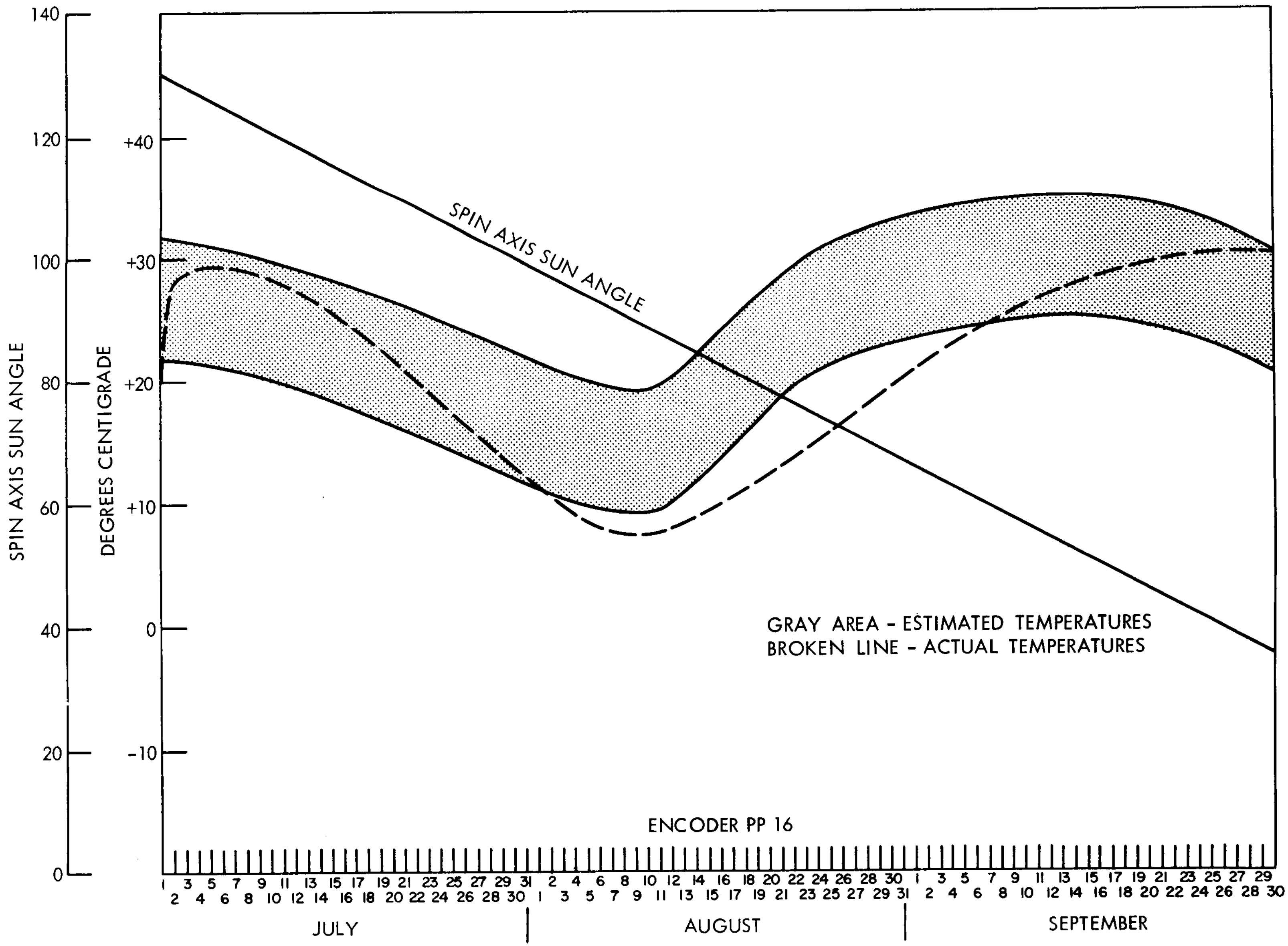


Figure 9–Encoder PP 16

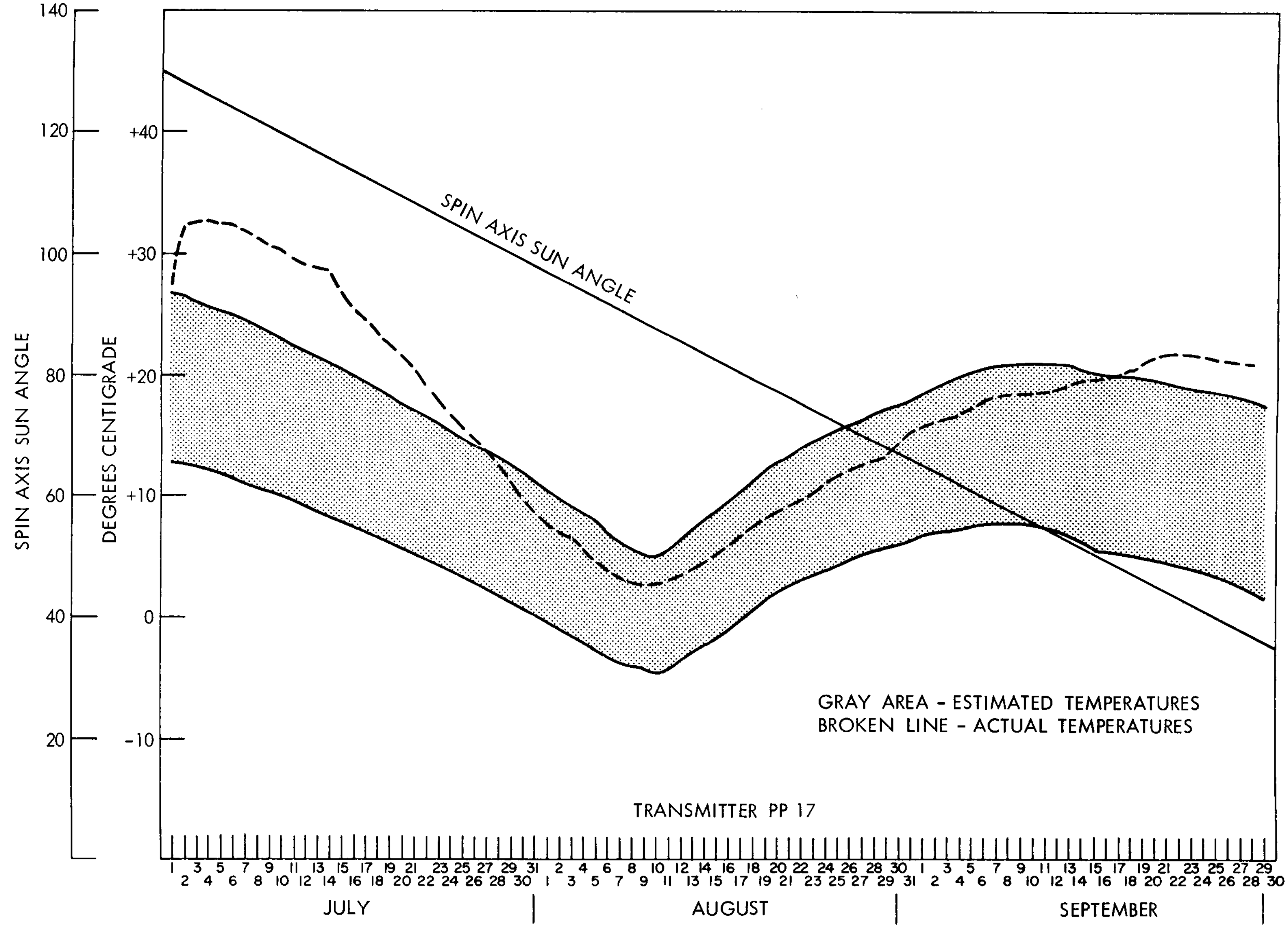


Figure 10–Transmitter PP 17

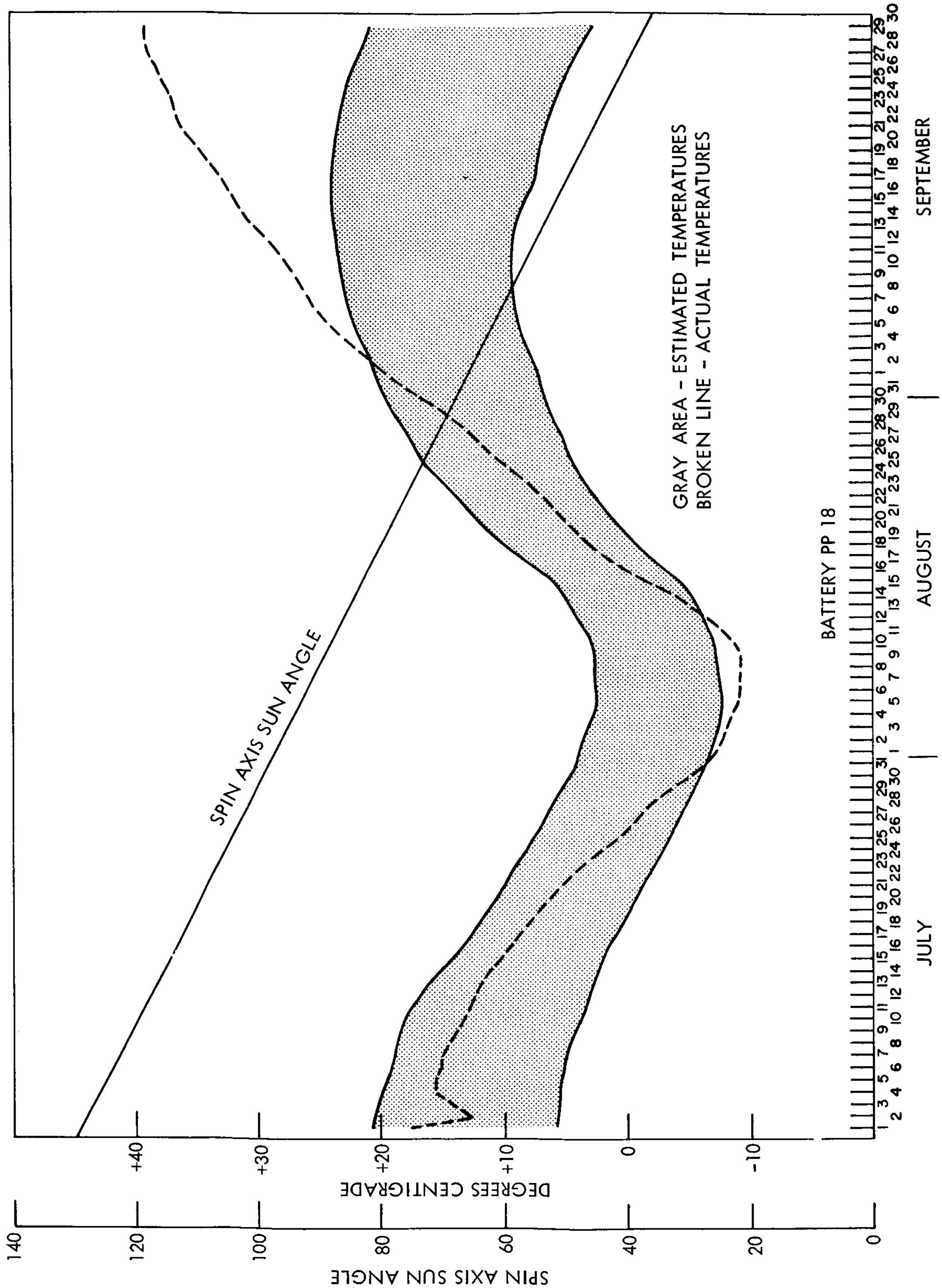


Figure 11—Battery PP 18

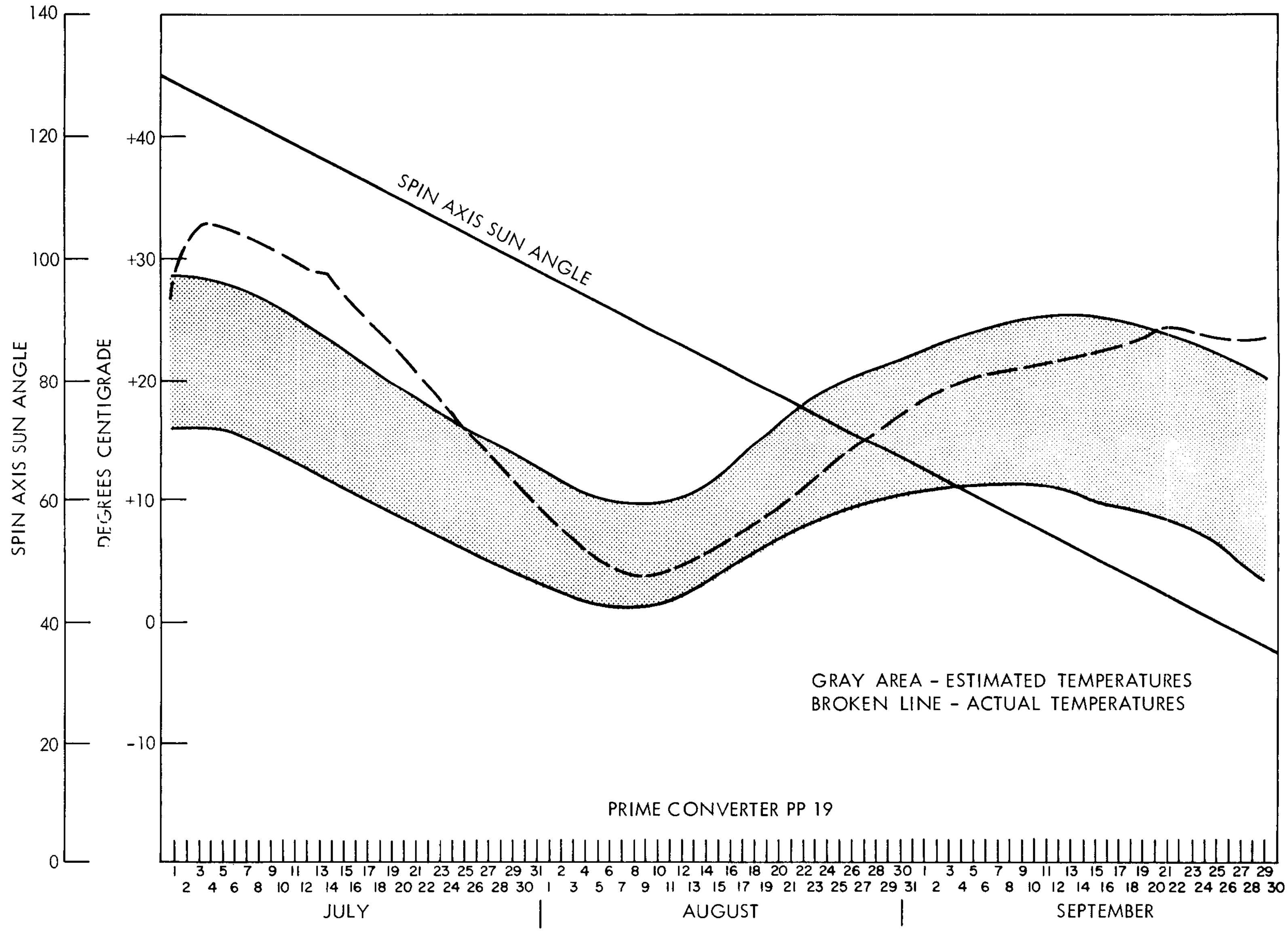


Figure 12–Prime Converter PP 19

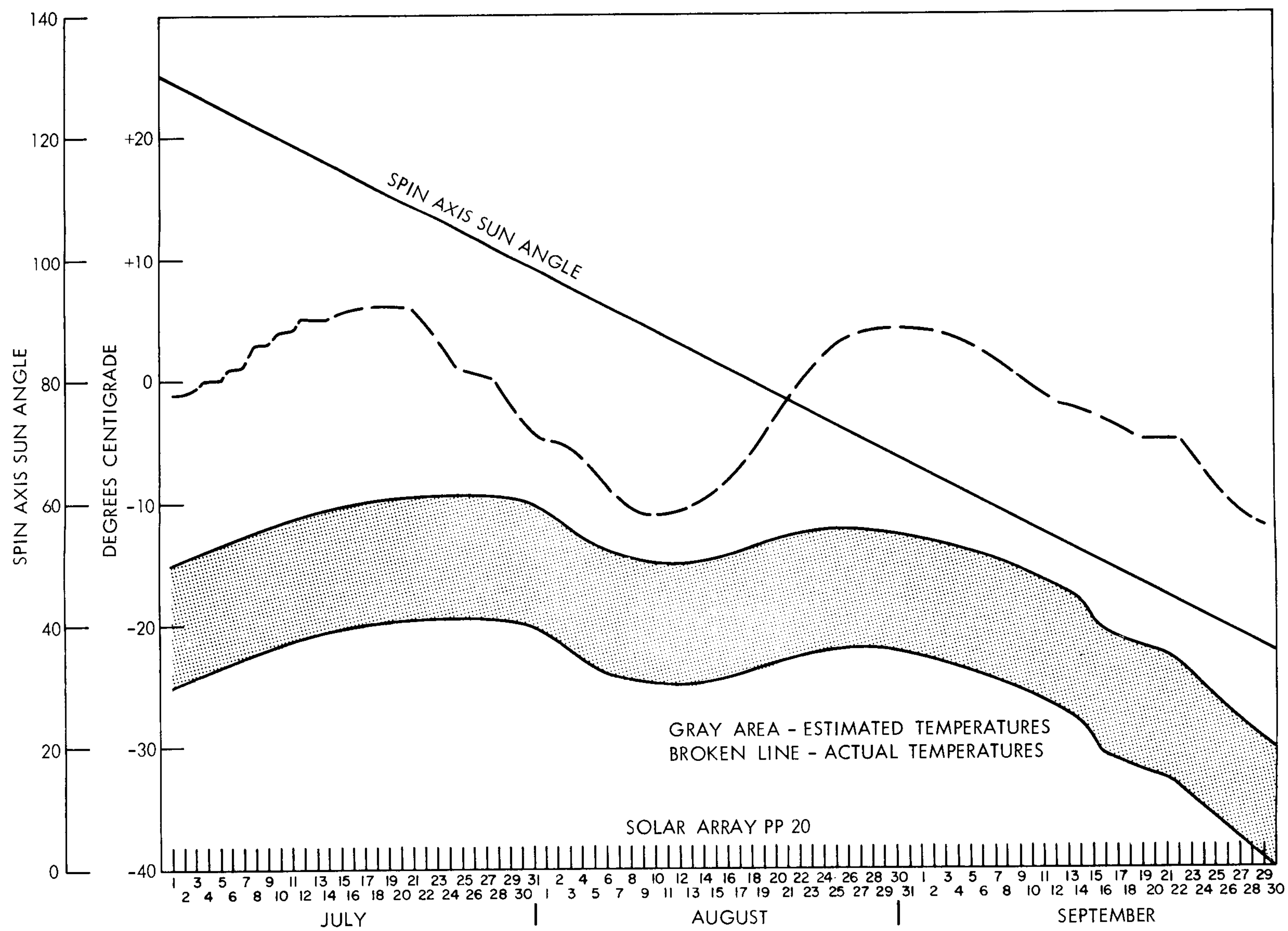


Figure 13–Solar Array PP 20

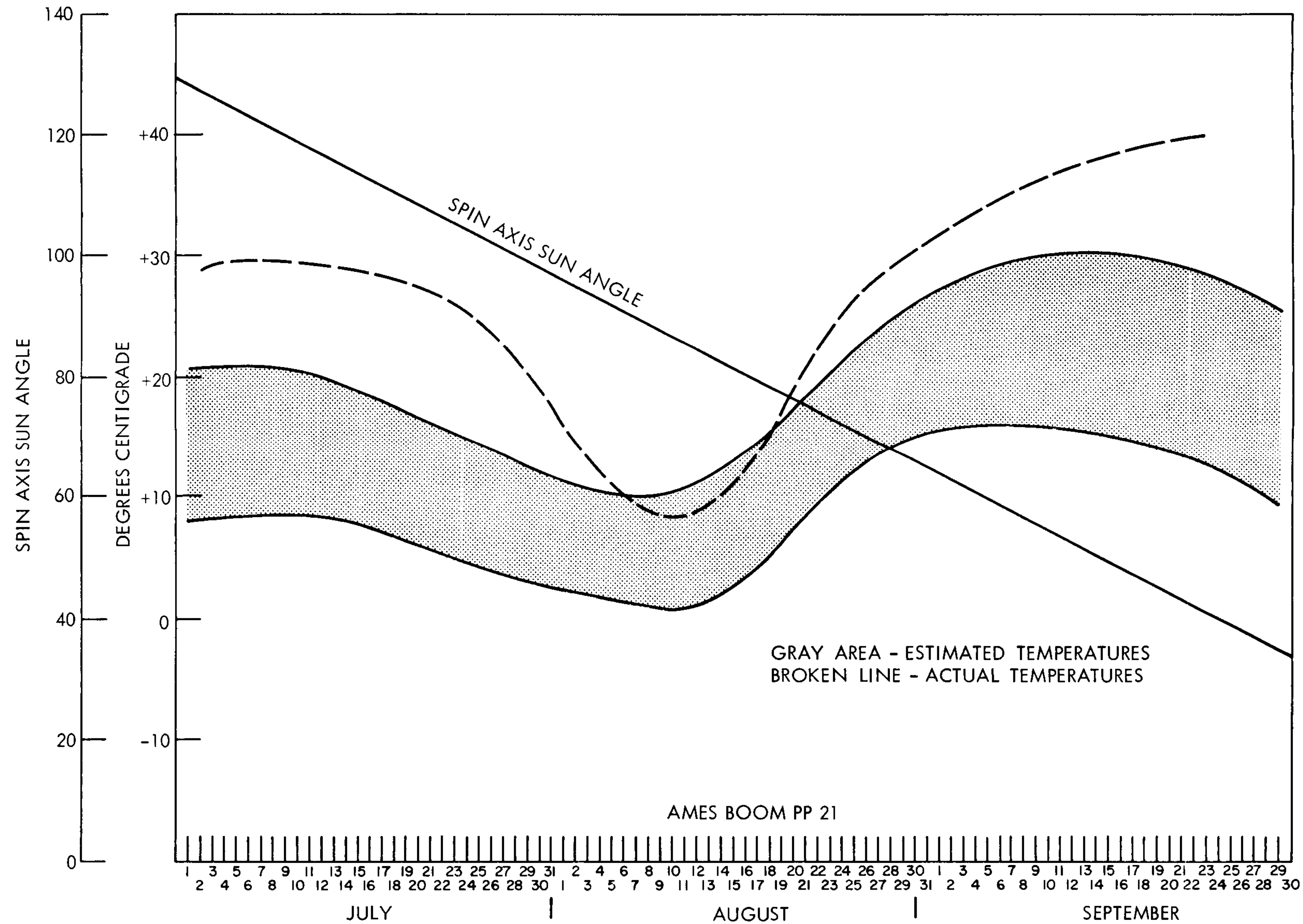


Figure 14–Ames Boom PP 21

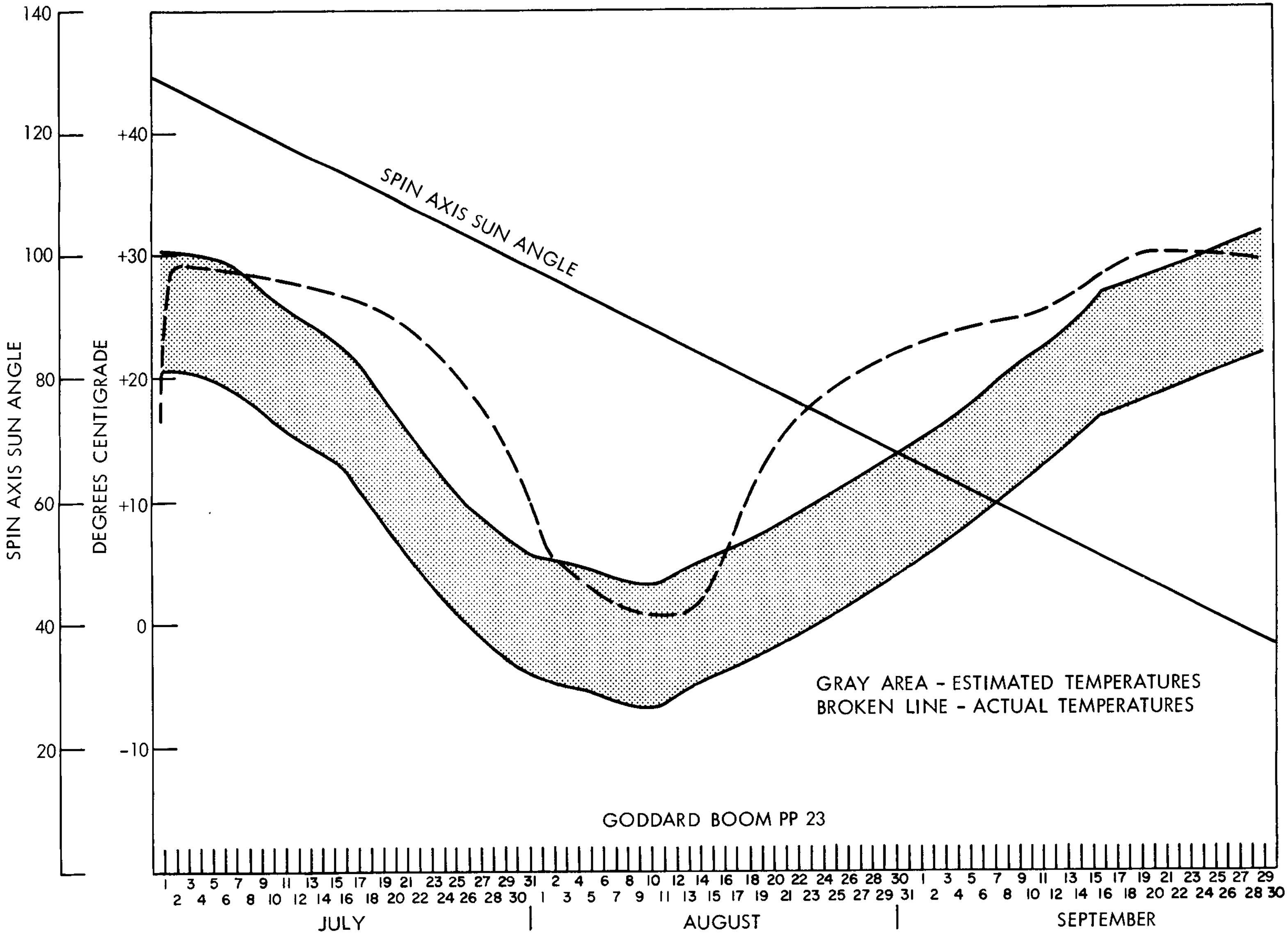


Figure 15–Goddard Boom PP 23

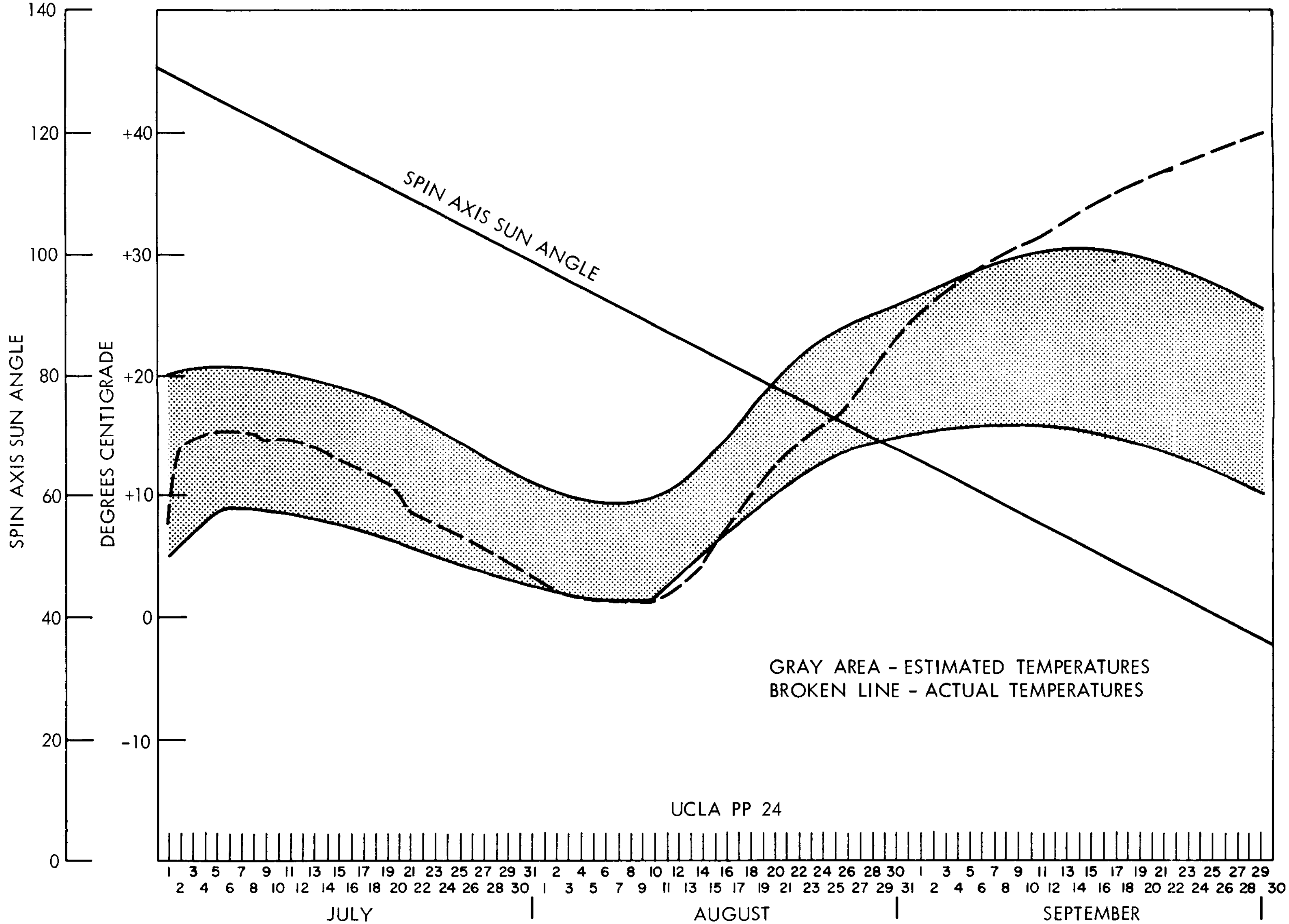


Figure 16–UCLA PP 24

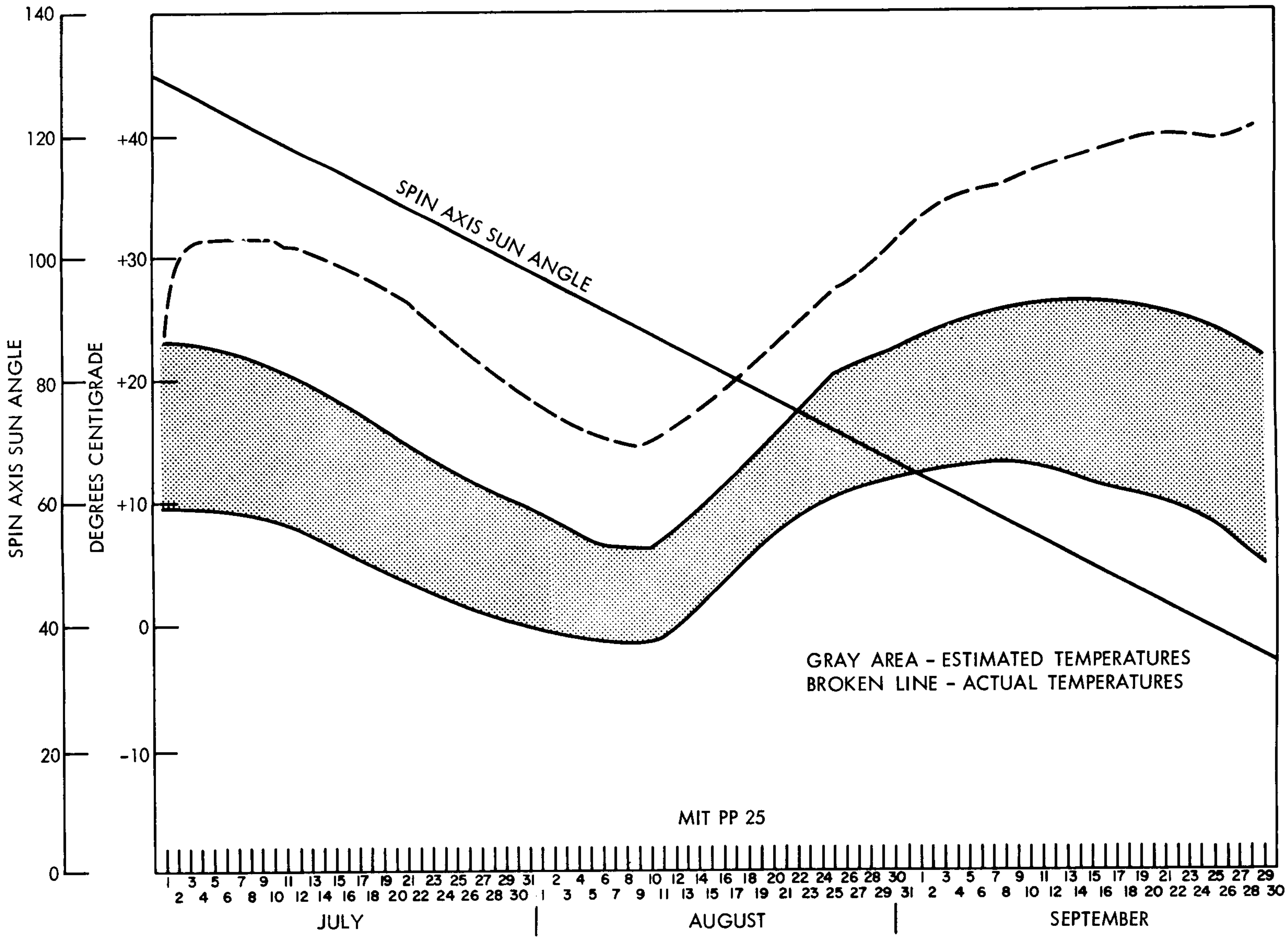


Figure 17–MIT PP 25

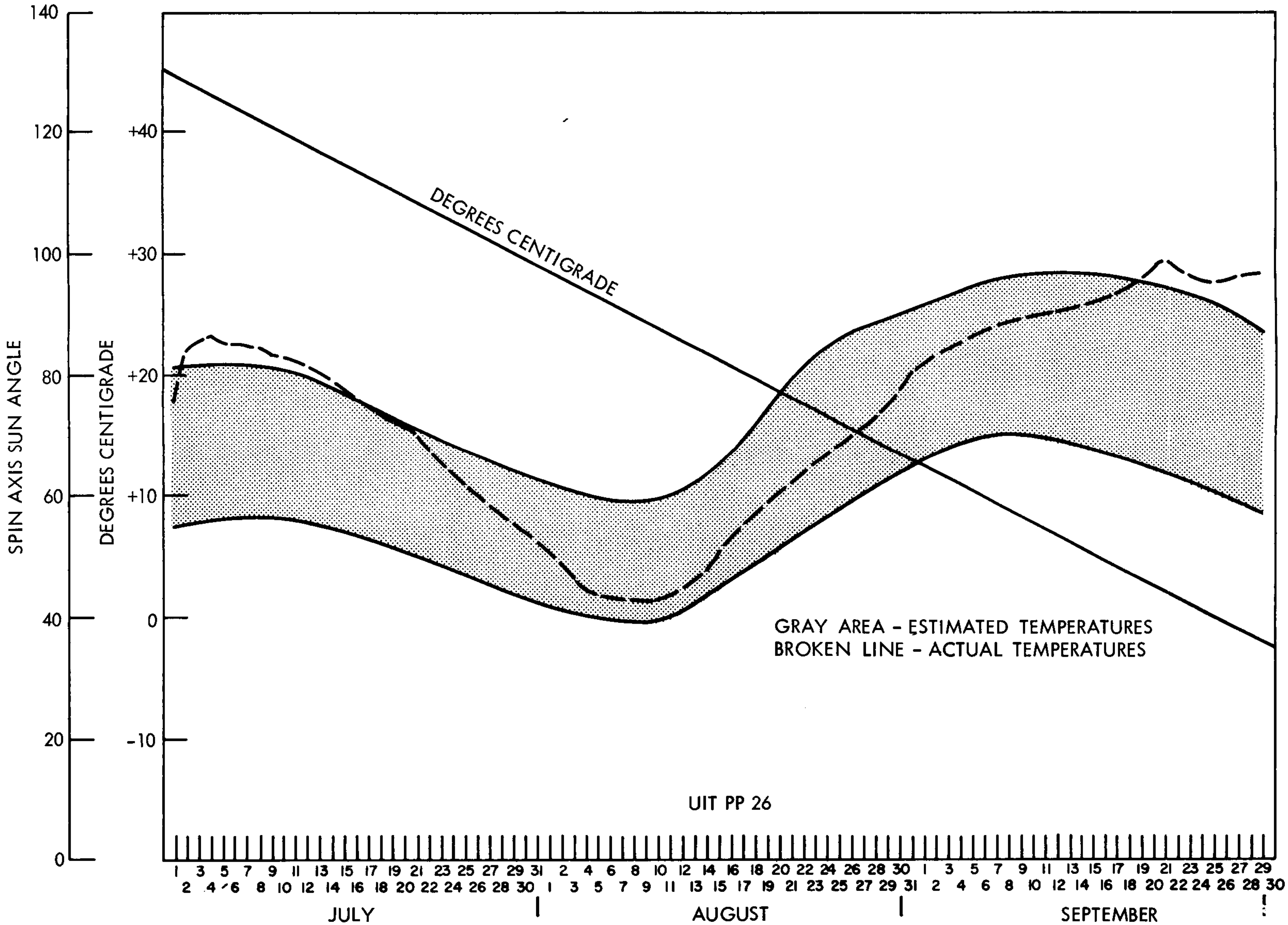


Figure 18–UIT PP 26

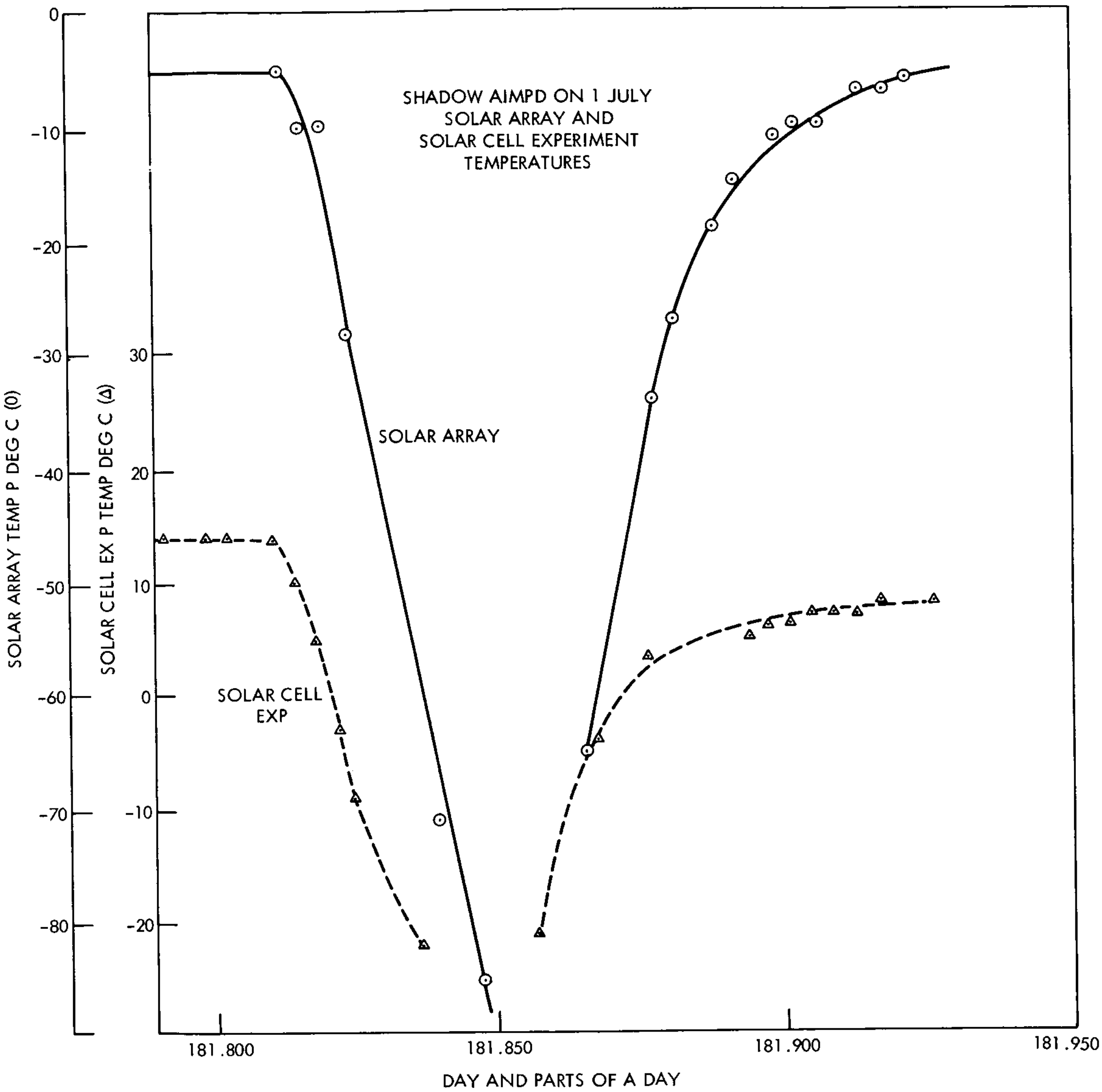


Figure 19–Shadow AIMPD on 1 July, Solar Array and Solar Cell Experiment Temperatures

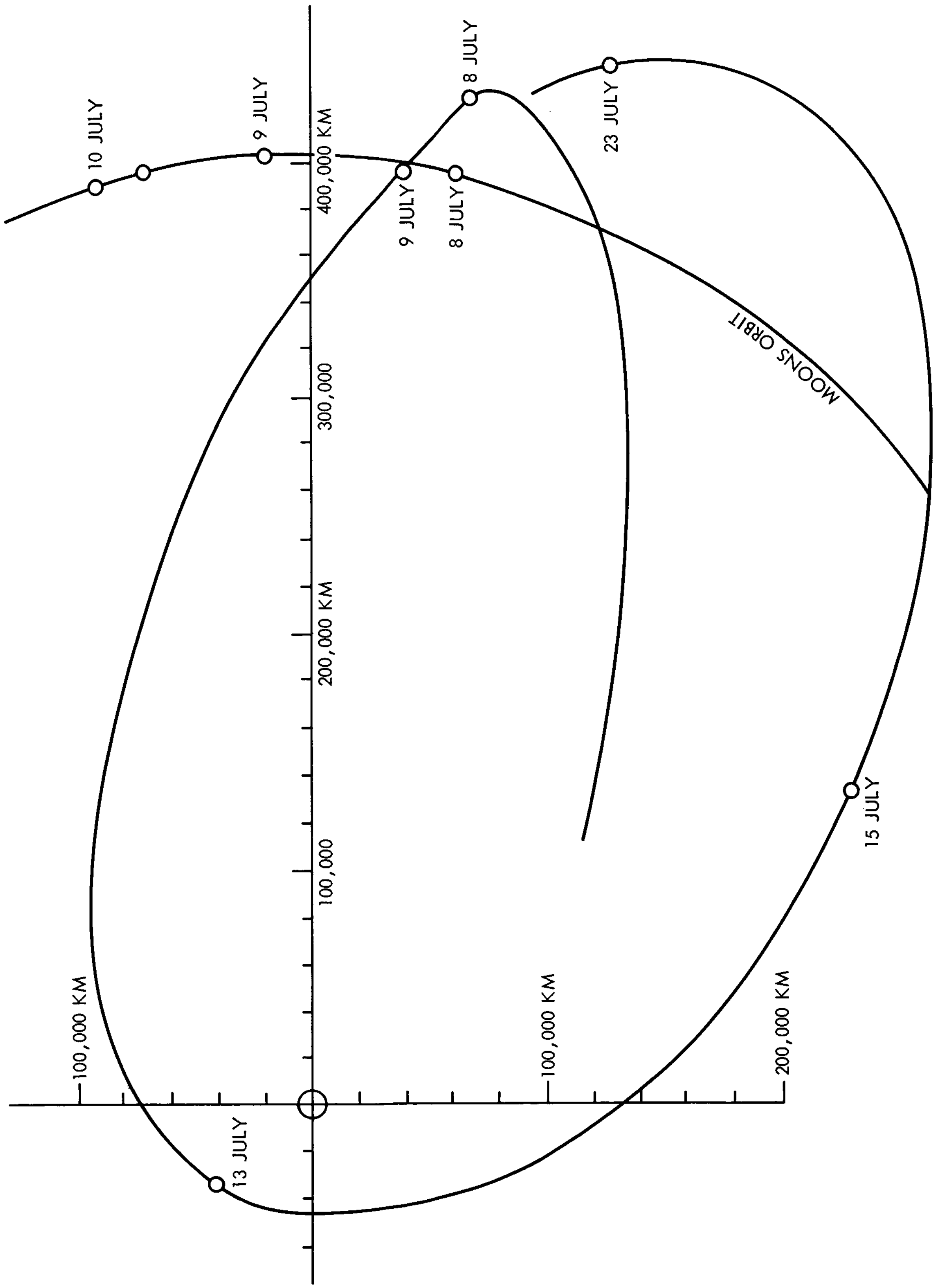

10 JULY
9 JULY
8 JULY
23 JULY
400,000 KM
9 JULY
8 JULY
MOONS ORBIT
300,000
200,000 KM
100,000
15 JULY
100,000 KM
100,000 KM
200,000 KM
13 JULY

Figure 20

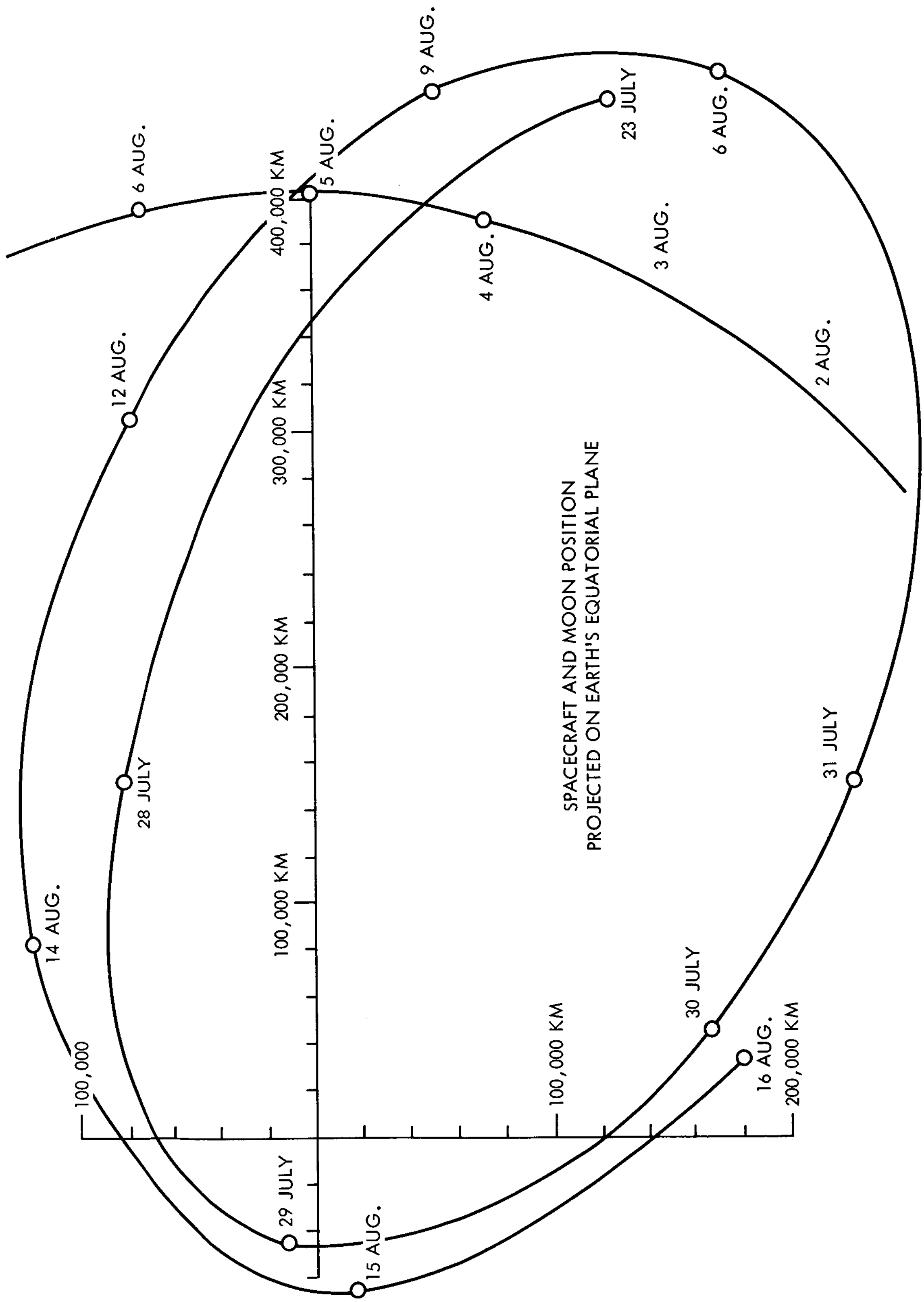

SPACECRAFT AND MOON POSITION
PROJECTED ON EARTH'S EQUATORIAL PLANE
100,000 KM
200,000 KM
300,000 KM
400,000 KM
100,000
100,000 KM
200,000 KM
23 JULY
28 JULY
29 JULY
30 JULY
31 JULY
2 AUG.
3 AUG.
4 AUG.
5 AUG.
6 AUG.
6 AUG.
9 AUG.
12 AUG.
14 AUG.
15 AUG.
16 AUG.

Figure 21

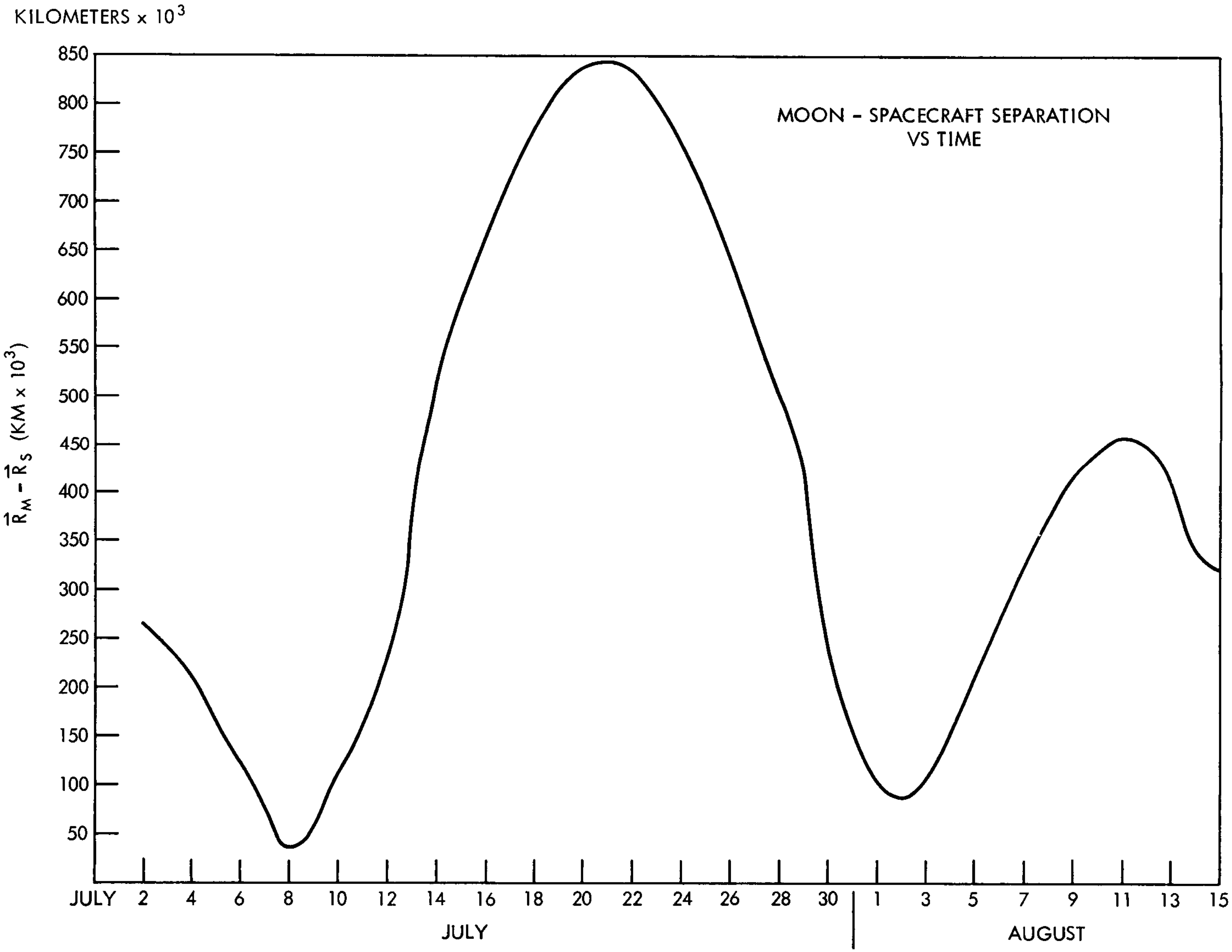


Figure 22—Moon-Spacecraft Separation vs Time

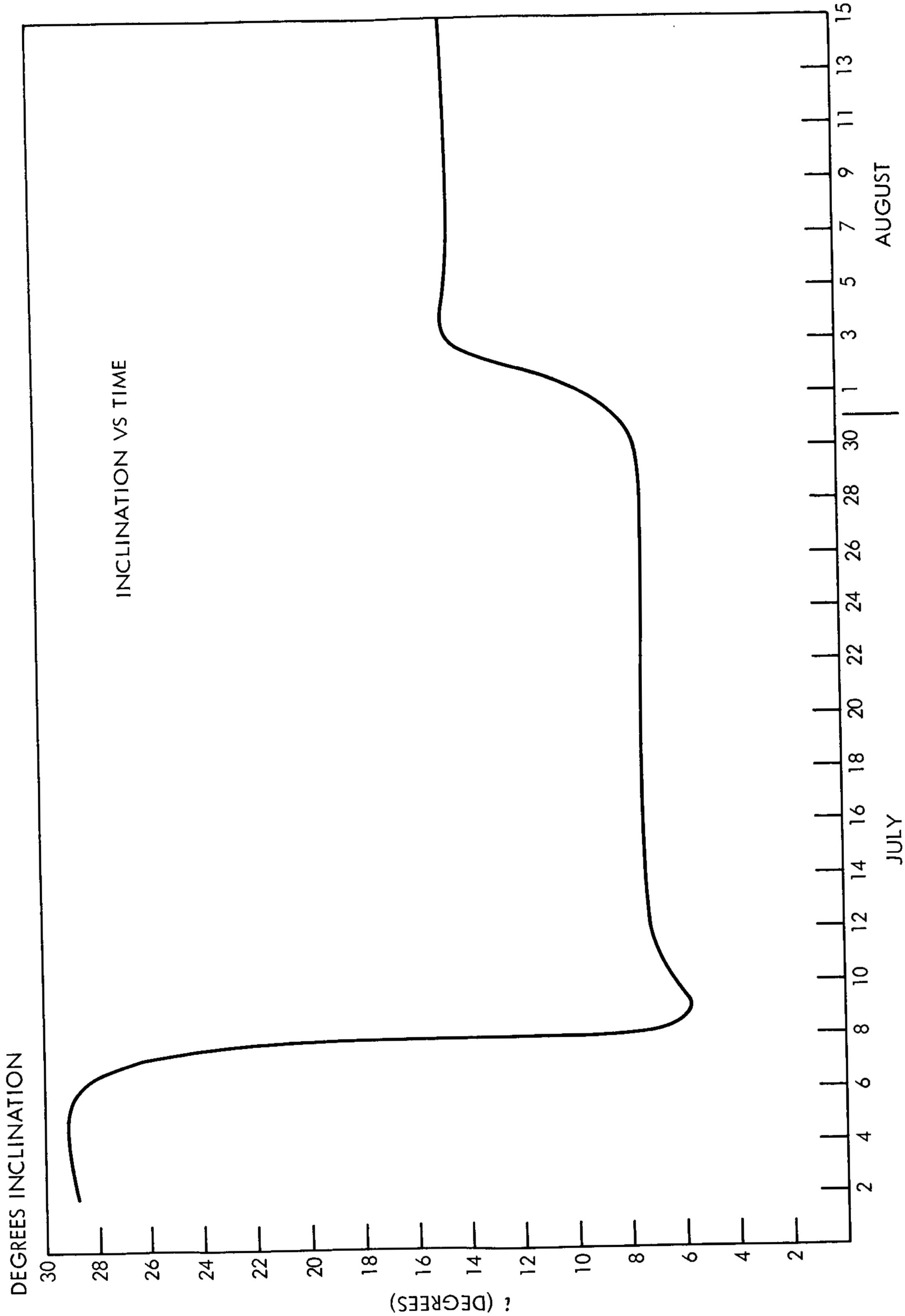
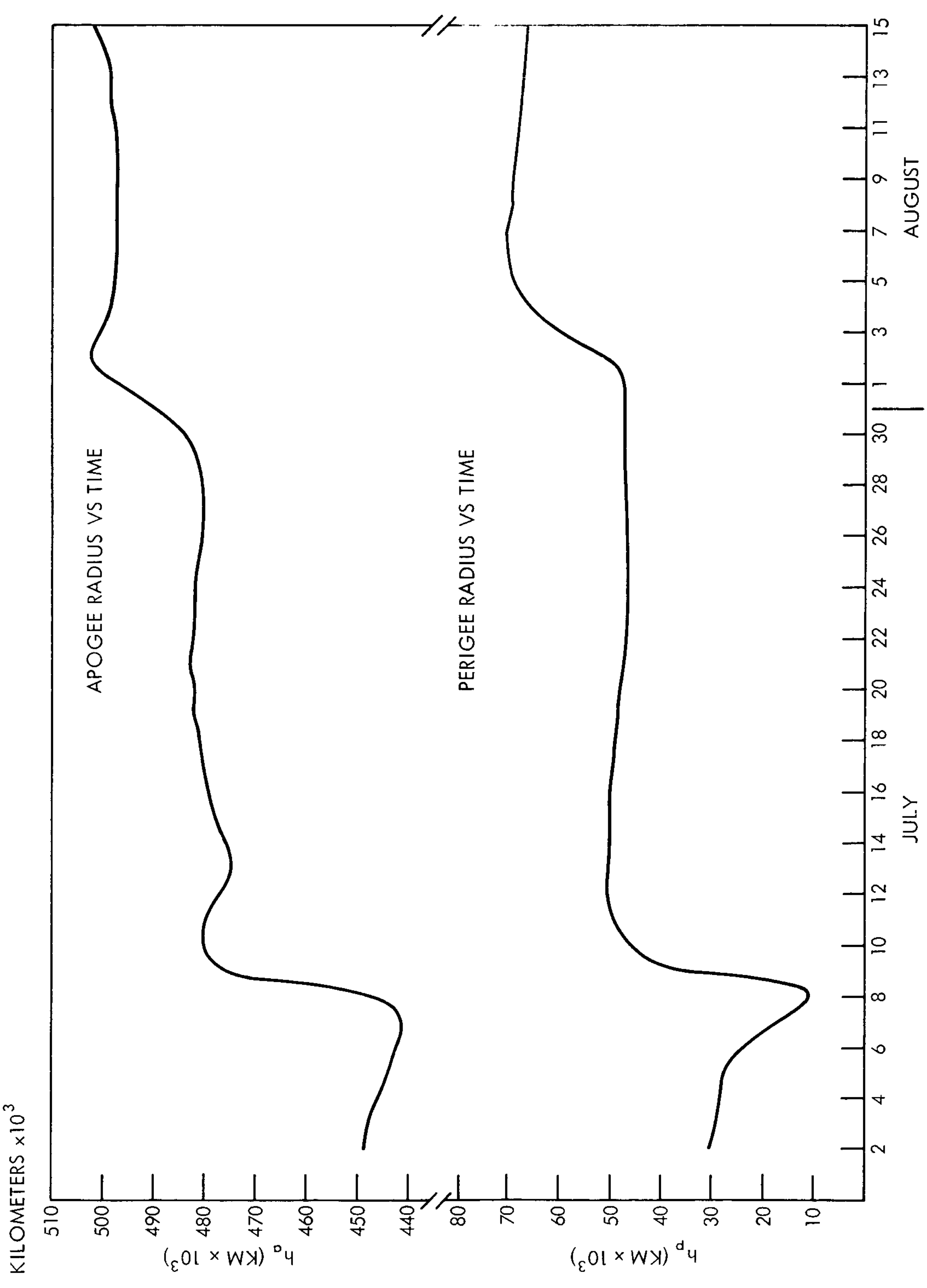


Figure 23–Inclination vs Time



KILOMETERS ×10^3
h_a (KM × 10^3)
h_p (KM × 10^3)
510
500
490
480
470
460
450
440
80
70
60
50
40
30
20
10
APOGEE RADIUS VS TIME
PERIGEE RADIUS VS TIME
2
4
6
8
10
12
14
16
18
20
22
24
26
28
30
1
3
5
7
9
11
13
15
JULY
AUGUST

Figure 24

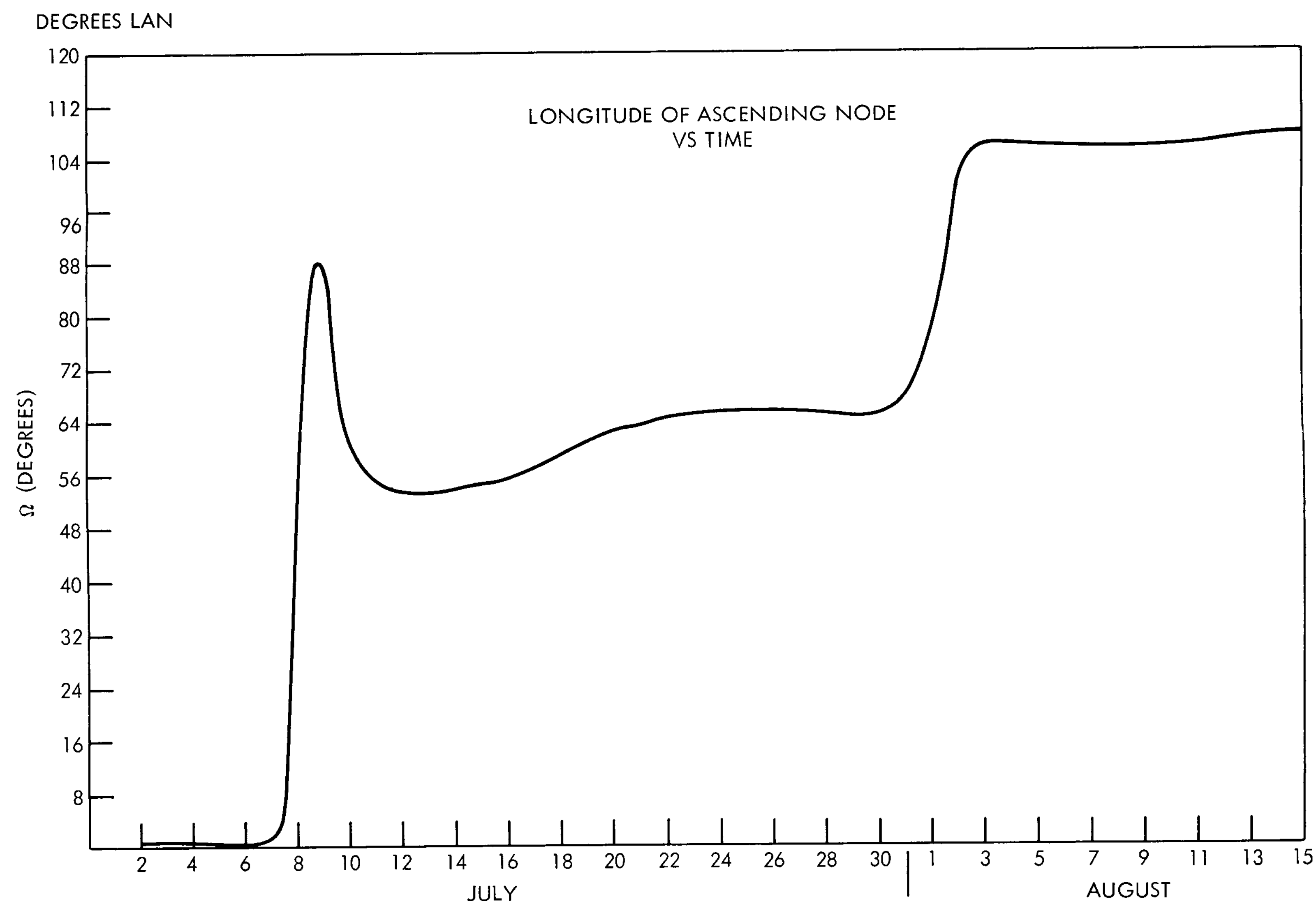


Figure 25–Longitude of Ascending Node vs Time

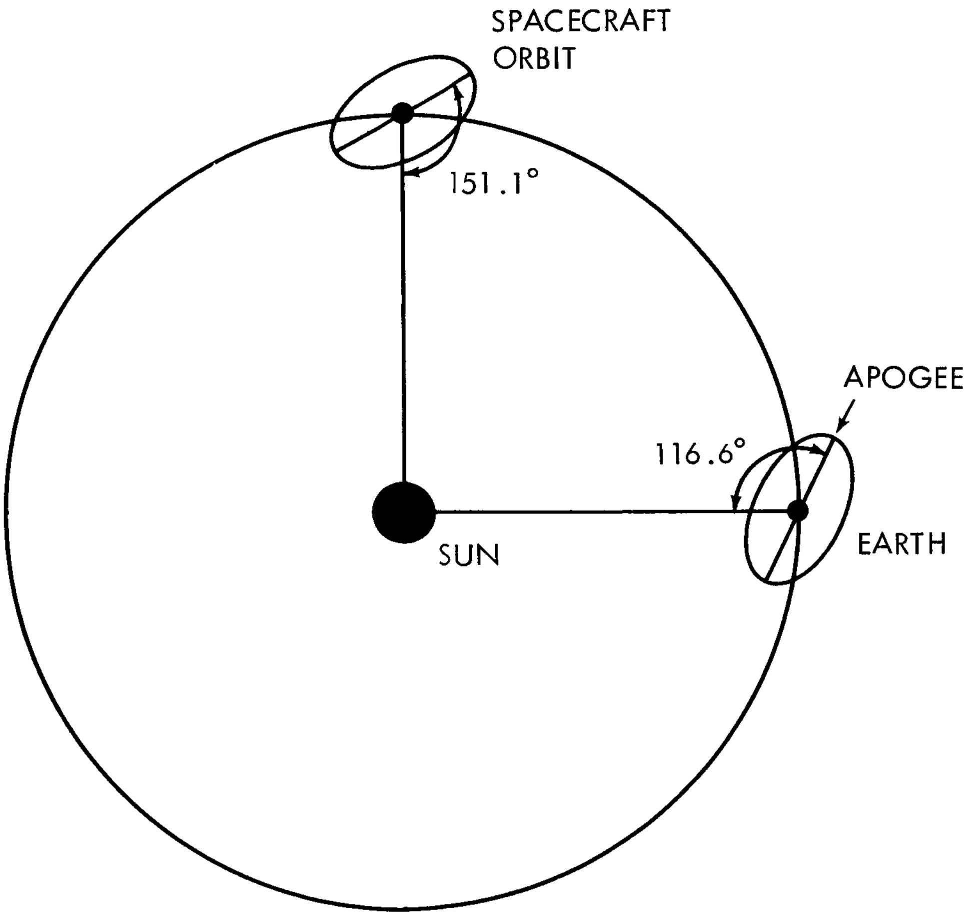


Figure 26–Line of Apsides - Sun Angle

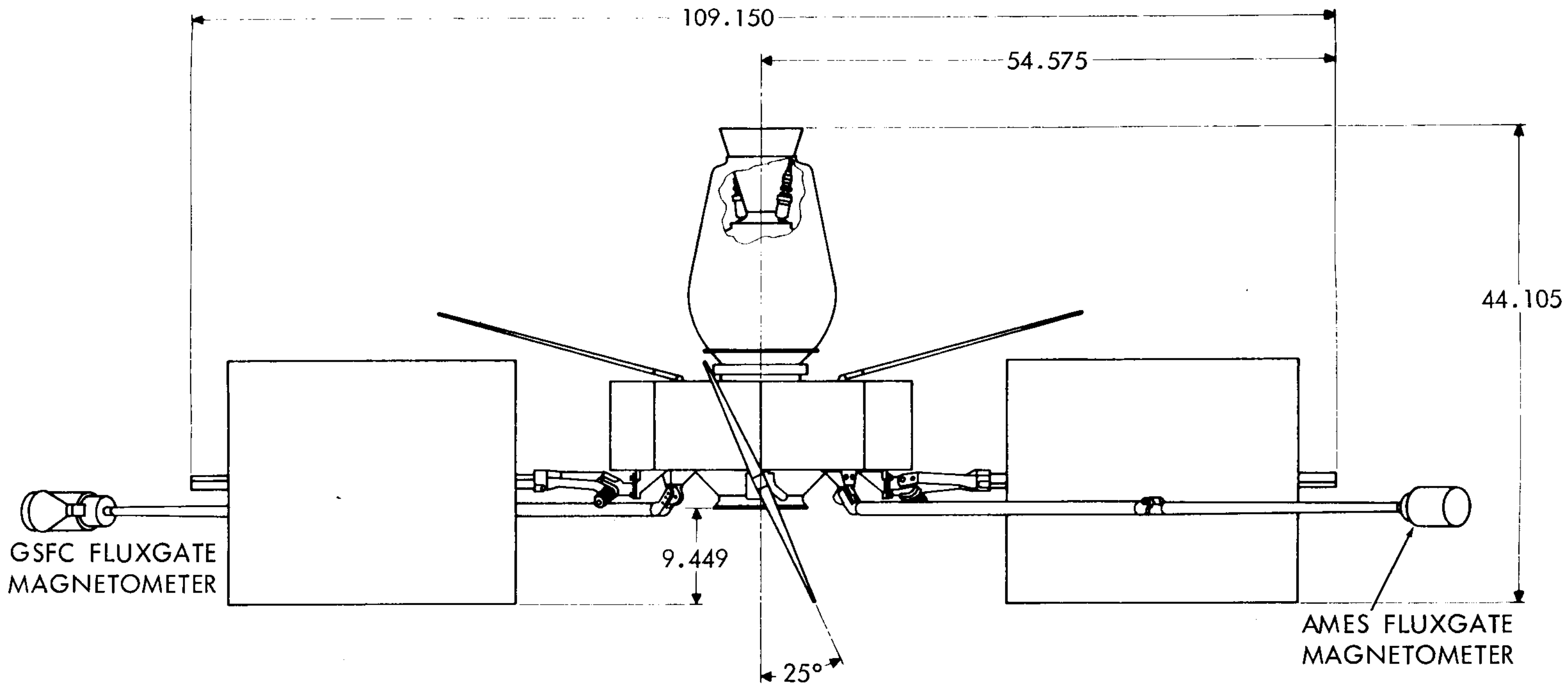


Figure A-1—AIMP-D Spacecraft Side View

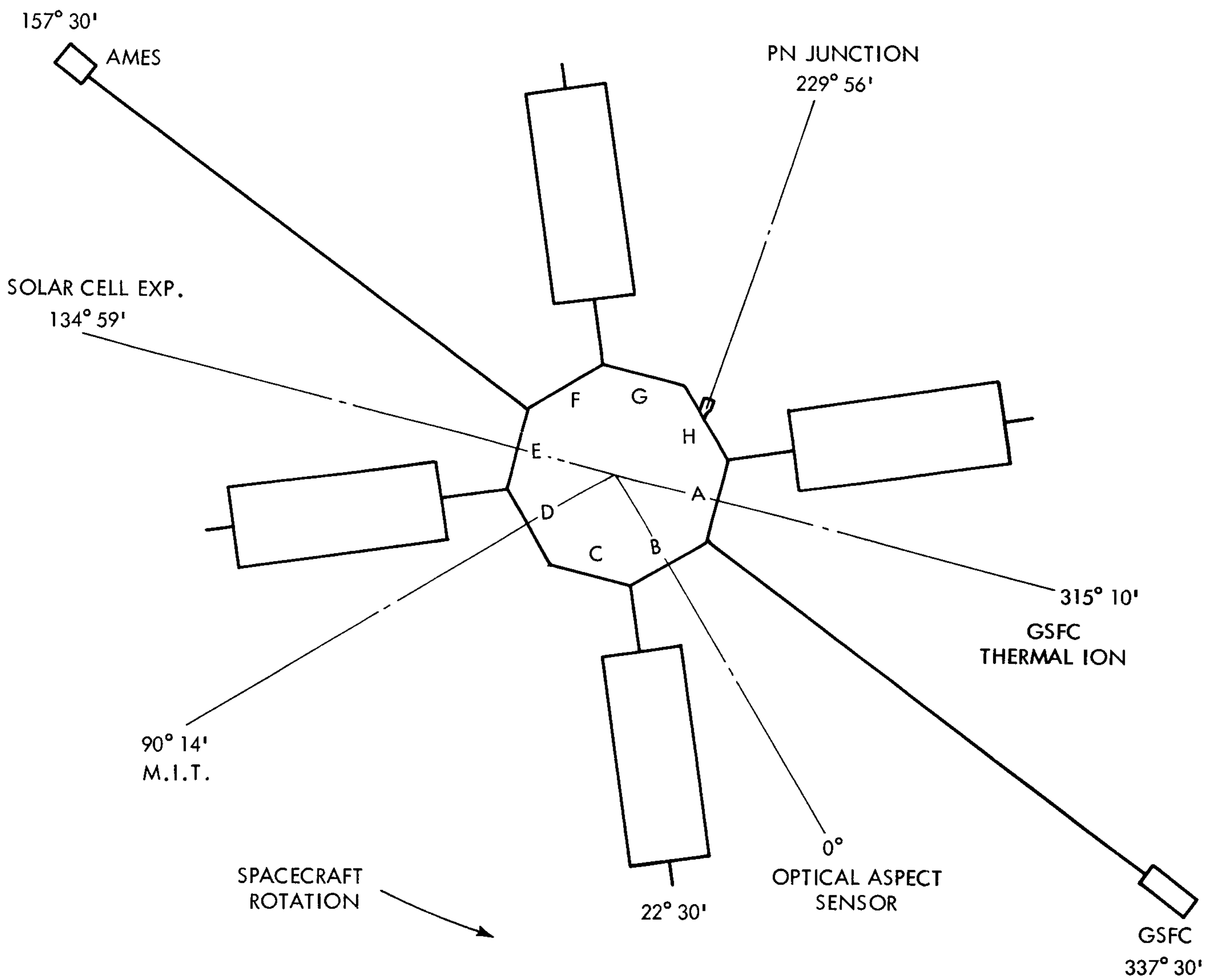


Figure A-2–Experiment Location Restreet to D.A. Sensor Iowa

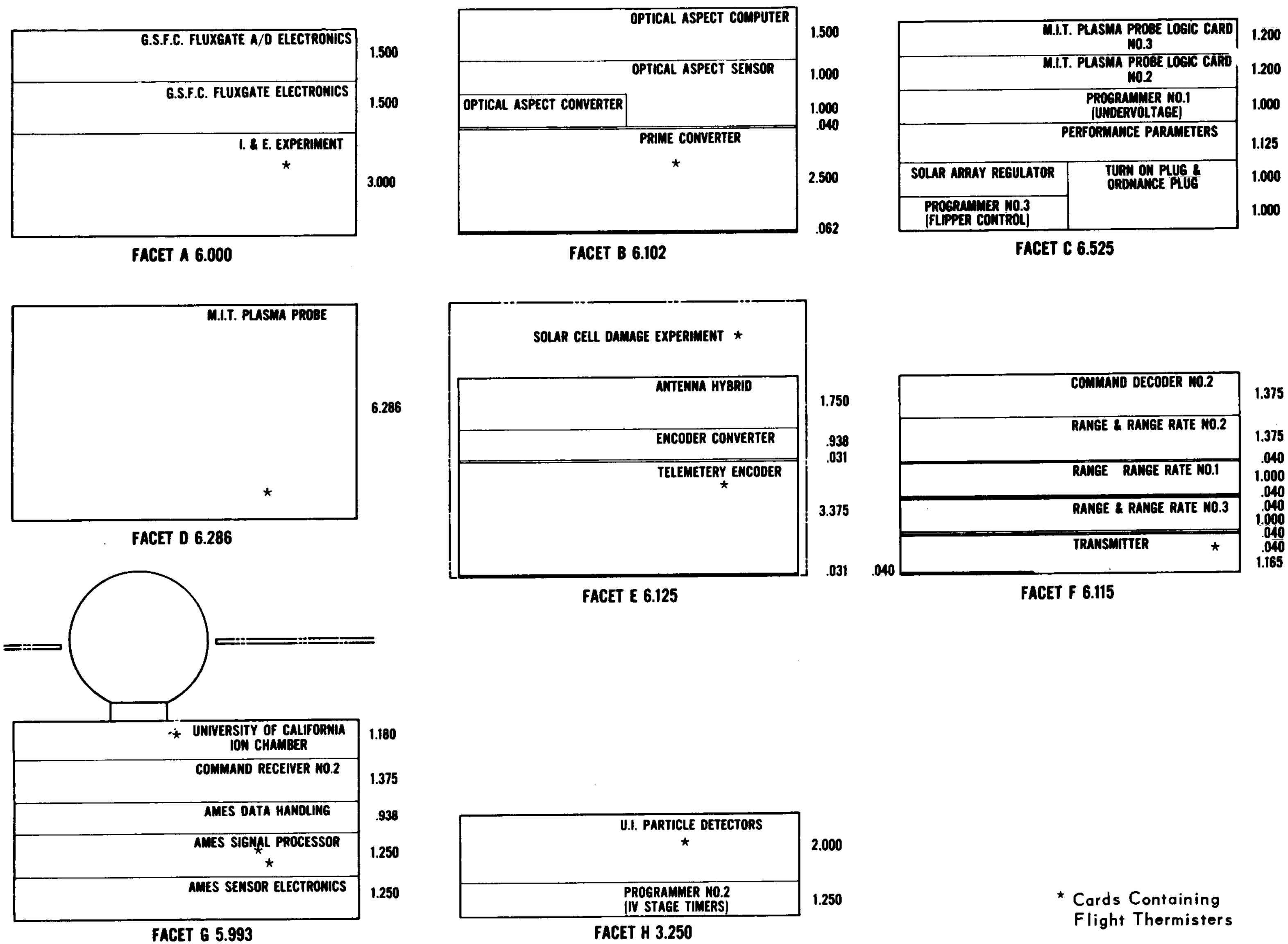


Figure A-3–AIMP-D & E Module Frame Location

X-724-66-588

INTERIM FLIGHT REPORT ANCHORED INTERPLANETARY MONITORING PLATFORM AIMP I - EXPLORER XXXIII

ERRATA

Page	Line	
2	10	"attenuation" vice "antenuation".
9	18	"excursion" vice "excersion".
9	30	"lifetime" vice "life time".
9	38	"evident" vice "evidenced".
10	14	"ascension" vice "ascention".
13	10	"shining" vice "shinning".
13	16	"micrometeorite" vice "micrometer".
13	17	"contamination" vice "decontamination".
25		The abscissa should read "Spin Rate in rpm".
39		"UCAL" vice "UCLA".
41		Slanted line should be marked "SPIN AXIS SUN ANGLE" vice "DEGREES CENTIGRADE".
43		Figure 20 should be labeled "First Spacecraft Orbit".
48		The abscissa should be labeled "DEGREES LON" vice "DEGREES LAN".
51		Figure title should be "Experiment Location Respect to O.A. Sensor" vice "Experiment Location Restreet to D.A. Sensor Iowa".
52		Bottom right hand corner of chart "Thermistors" vice "Thermisters".